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A REVIEW OF FULL-SCALE TESTING OF AIRCRAFT STRUCTURES AT THE NATIONAL AERONAUTICAL ESTABLISHMENT

by

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A HISTORY OF FULL-SCALE TESTING OF AIRCRAFT STRUCTURES AT THE NATIONAL AERONAUTICAL ESTABLISHMENT

L'HISTOIRE DE L'ÉVALUATION EN LABORATOIRE DE STRUCTURES D'AÉRONEFS À L'ÉTABLISSEMENT AÉRONAUTIQUE NATIONAL

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SUMMARY

This report presents an historical review of the work of the Structures and Materials Laboratory of NAE in the area of full scale testing of large aircraft structures. It covers the period from 1941 to 1984, starting with a static strength test of a moulded wood Anson fuselage and finishing with a fatigue test of a Grumman Tracker wing. Brief details of the loading arrangements and test results are included for each test component and these are used to trace the development of the laboratory from the use of rulers and shot bags to computer-controlled servo-hydraulic actuators.

RÉSUMÉ

Le présent rapport fait une récapitulation historique du travail du Laboratoire des structures et des matériaux de l'ÉAN dans le domaine de l'évaluation en laboratoire de structures d'aéronefs. Il couvre la période allant de 1941 à 1984, en commençant par l'essai de résistance statique d'un fuselage Anson en bois et en terminant pas un essai de fatigue de l'aile d'un Grumman Tracker. De brefs détails sur les conditions de charge et les résultats des essais sont également inclus pour chaque essai, et ces détails servent à retracer les progrès au laboratoire depuis l'emploi de règles à mesurer et de sacs de grenaille de plomb à l'emploi de verins servo-hydrauliques commandés par ordinateur.

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A HISTORY OF FULL-SCALE TESTING OF AIRCRAFT STRUCTURES AT THE NATIONAL AERONAUTICAL ESTABLISHMENT

INTRODUCTION

The conduct of full-scale structural fatigue tests is both time consuming and expensive and requires considerable expertise in all areas of structural fatigue including loads monitoring, structural testing, crack propagation analysis, non-destructive testing etc. Because of its mandate and areas of interest the Structures and Materials Laboratory of NAE is able to maintain a pool of this expertise which is available, together with the associated experimental facilities, for outside agencies when and as required.

As part of a larger review of current facilities and capabilities in the light of new developments in both areas, a review of the past work of the laboratory was undertaken and is presented here. In general, only large structures have been included. Many smaller components have been tested over the years but these were considered beyond the scope of this review. Similarly, considerable work has been undertaken within the laboratory on the impact resistance of components and structures, but this is also beyond the scope of this paper.

STATIC TEST OF MOULDED WOOD ANSON FUSELAGE

One of the earliest full scale tests on an aircraft structure performed by this laboratory, which was then the Structures Laboratory of the Division of Mechanical Engineering, was a static test of a moulded wood Anson aircraft fuselage[1] in 1941. The test was conducted at the Vidal Research Corporation, New Rochelle, N.Y. for the Royal Canadian Air Force (RCAF). The fuselage was fitted with dummy mainplane spars, which were tied down to heavy steel girders laid on the floor and braced to the structural steel of the building. Loading was carried out by means of shot bags for the down-loads and by a hydraulic jack and platform scale with a steel girder multiplying lever for the tail-skid up-load. Loading was applied on the floor boards of the fuselage, on a temporary frame on top of the fuselage and, for side loading, via a horizontal cable from a tubular member carried in two wooden frames fitted to the fin, and then over a large diameter pulley to a weight platform. Deflections were measured by reading scales on the fuselage side with a surveyor's level or, for side loading, by attaching scales to the fuselage with double sighting wires, to eliminate parallax, attached to independent structures.

The structure was loaded to 75% of ultimate load at various intervals in both side and down loading and deflections noted. There were no indications of creep or permanent deformation. The fuselage was then tested to destruction for the down load without nose load as this was observed to be the most critical loading case. At 140% ultimate load a compression failure occurred just aft of the door. The report concluded that the fuselage had an adequate margin of strength and that this could be improved by judicious rearrangement of the weight of material used and some modification of the structure to obtain a better stress distribution.

It is also interesting to note that the report recommended that the fuselage be made available for weathering tests since it was without surface finish of any kind.

TIGER MOTH MAINPLANES

One of the first tests within the laboratory was a static test of four upper mainplanes of Tiger Moth Aeroplanes for the National Defence for Air in 1942[2]. The object of these tests was to determine to what extent the mainplanes were deteriorating in service due to doubtful gluing, and wings having 0, 400, 800 and 1000 flying hours were tested.

They were rigged in the inverted position in a test stand as shown in Figures 1 and 2 with root end fittings provided to apply the correct type of reaction. Slot and balancing loads were provided by loading platforms and a system of wooden levers (early whiffletrees) while a box on the lower wing was located to provide strut reactions. Short slats spanned the rib intervals and transmitted shot bag loads directly to the rib trusses. A frame supported by hydraulic jacks provided an abutment for the structure to rest on during loading operations, while a sling was loosely looped around the lower wing as a precautionary measure. Deflections along the leading and trailing edges were measured by a mechanical "strain gauge" reading to 0.001 ft. All wings were loaded to 120% of ultimate load without collapse and it was concluded that there was no evidence of deterioration with age or failure in the glued joints.

HARVARD II MAINPLANES

The first tests of an all metal structure appear to have been in 1942 also, when two Harvard II Starboard Mainplanes were tested for National Defence, Air Service[3]. These tests were to observe under what conditions the wings, which were stressed skin structures, showed evidence of wrinkling. The angles at the wing root were bolted to a face plate and the wing mounted in the inverted position and loaded up to 90% of the specified ultimate load via shot bags with the wing tip supported by a beam suspended from a crane during loading operations as shown in Figure 3. Ailerons were not used in the test and appropriate loads were applied directly to the hinge brackets. Deflections were again measured by mechanical "strain gauges". It was concluded that the observed wrinkling shown in Figure 4 was normal for a stressed skin structure.

MOULDED WOOD HURRICANE WING PANEL

Another type of structure was tested in 1942. This was a moulded wood Hurricane wing test specimen tested for the British Purchasing Commission[4]. The test specimen was designed and constructed by Vidal Research Corporation and represented in simplified form the outer ten feet of a forty foot wing span of a "modern fighter type aeroplane". The wing was of a plywood, stressed skin — lattice construction and consisted of four large veneer flanges tapering in section and moulded into the skin, a channel section shear web and a moulded plywood covering. The remainder of the stabilizing structure comprised a series of geodetic stiffeners inside the skin spaced by tubular aluminum struts. The specimen did not perform well when loaded in the normal way using lead shot bags, and failures occurred at 50% of the specified test load. Figure 5 shows the test section before loading with the 'strain gauges' attached to the leading edge.

MOULDED PLYWOOD HARVARD FUSELAGE

At the beginning of the war a program had been instituted to develop and encourage the use of wood and plywood in aeroplane construction in case there was a shortage of aluminum. Several projects were undertaken and in late 1941 work had begun on a moulded plywood version of the rear fuselage and tail surfaces of the Harvard advanced trainer aeroplane. These were tested in the laboratory in late 1942[5].

The fuselage skin was of moulded spruce plywood, formed and bonded by the bag moulding process. Vertical frames of light multi-ply construction as well as four stub longerons in the forward section and four stringers in the mid section completed the framework. The horizontal stabilizer, elevator and rudder were also constructed of moulded plywood. Critical loading conditions for the portion of the structure tested were obtained from the type record and all loads were applied for a minimum of one minute.

The fuselage was fitted to a tubular test jig that approximated the rigidity of the rear section of a welded tubular mid section of the Harvard. This jig was then supported on a welded 12 inch I-beam structure. The fuselage was mounted in both upright (Figure 6) and horizontal

positions (Figure 7) for various loading conditions and loads applied by means of lead weights and lead shot bags for down loads and by a hand hoist, scale balance and a steel I-beam for up loads. Deflections were measured with a surveyors level and "strain gauges". All the design loads were successfully carried except for a "special tailwheel condition B" which resulted in a failure of the lower skin fibres at 100% design load, Figure 8, due to a stress concentration at the sharp corner of the flare cut-out and lack of continuity of the short stiffeners which carried the flare tube brackets. After repair and modification the design load was successfully carried.

The stabilizer was mounted in a horizontal position and one third of the load applied to the upper skin by shot bags, the remainder being applied by means of load pans and a linkage arrangement at seven stations spanwise and four chordwise, woodscrews being utilized to apply the force to the skin at the stiffener, Figure 9. The elevator was mounted in a horizontal position and loaded in a similar manner to the stabilizer, Figure 10.

Since these tests were all satisfactory, a second unit was prepared for flight testing in September 1943 and after preliminary flight trials at the RCAF Test and Development Establishment, Rockcliffe, was assigned to routine flying at Uplands Air Station. The aircraft was later returned to NRC and assigned to test flying in connection with a stability investigation.

ANSON MAINPLANES AND TAILPLANES

In 1943 static tests were conducted on four Anson mainplanes for the Department of National Defence for Air[6] to determine the reduction in strength resulting from glue deterioration in the spars of two condemned Anson I mainplanes and to determine the strength of a new Canadian built mainplane for use in the Anson V aircraft. The 56.5 ft mainplane was loaded in the inverted position using slats and shot bags in the normal way as shown in Figure 11 while engine and undercarriage loads were provided by a crane reaction through a whiffletree arrangement shown in Figure 12. Deflections were measured by the same mechanical "strain gauges" referred to above, although in this report they are called "tape strain gauges", while dial gauges were used to monitor the deflection of the test stand. The condemned mainplanes, one classed as in "very bad condition" (and labelled "write-off") and one in "fairly bad condition" failed at 75% and 90% respectively of the design loading while the new mainplane failed at 95% design load. It was concluded after observing the failures that while the glue in the old spars (a cold setting urea formaldehyde) had deteriorated to some extent, the mainplanes had not failed as a result of this deterioration, but had failed like the new one at the centre section of the tension spar where a 5/8 inch bolt hole reduced the section. As a result of this observation, a further condemned mainplane in a similar condition to the previous two was modified by the addition of a 5/8 inch birch lamination in the critical area. This modified wing failed at 105% of the design load at a different location.

Similar tests were performed on Anson tailplanes in the same year[7]. Four tailplanes labelled "bad case of glue failure", "unserviceable, repairable", "unserviceable" and "failed on glue test" were found to sustain over 200% of the design load although it was observed that the glue in the spars had deteriorated. It was concluded that this did not affect their strength, however, since they failed in simple bending. A test set-up for a tailplane is shown in Figure 13. An interesting aspect of these last two reports[6,7] is that each contained full details of the load calculations.

DROP TEST OF A CORNELL AIRCRAFT

A different kind of full scale aircraft test was performed in 1944 when a drop test of a complete Cornell aircraft was carried out for the Department of National Defence for Air[8] to determine whether landing gear loads might produce centre section spar damage which might later contribute to structural failures in the air. This investigation was a result of several failures of Cornell wings in flight.

The aircraft tested was a Mark II Cornell made in Canada by Fleet Aircraft with less than 13 hours flying time. It was loaded to its maximum gross weight with lead shot bags. The rear of the fuselage was pivotted on the floor by anchoring the tail wheel assembly and the aircraft lifted with a chain hoist with a quick release mechanism attached to the slinging eye on the turnover structure as shown in Figure 14. The drop height was measured from the floor to the bottom of the tire with the landing gear extended to its extreme position. Tests were started with a 12 inch drop, then an 18 inch drop followed by 3 inch increments until some failure occurred in the aircraft. The tires were arrested by a concrete floor in the three point landing condition while for the drift condition they were arrested by two heavy timber ramps inclined at 15° to the horizontal to provide some side load on the undercarriage. Vertical accelerations were measured in the three point landing condition by recording accelerometers and the tests were photographed at 64 frames/second on 16mm film.

In the three point landing tests, minor damage occurred after the 30 inch drop corresponding to about 4-1/2g and the centre section wing skin failed after the 39 inch drop corresponding to about 6-1/2g.

After replacing the damaged centre section, the side drift landing tests were conducted which resulted in a skin failure after a 21 inch drop. This was attributed to a malfunction of the landing gear. After repairing the skin and replacing part of the landing gear, the skin showed no failure until after the 30 inch drop. It was concluded that a landing shock of 6 to 7g was required to fail the centre section skin.

Following on from these tests, static tests were performed on a Cornell wing[9] again for the Department of National Defence Air. A complete wooden cantilever wing consisting of a centre section and removable outboard panels as shown in Figure 15 was loaded in the inverted position via shot bags on short slats spanning the rib intervals. During the loading operation the wing tip was supported by jacks. Deflections were measured by "tape strain gauges" and the test stand deflections checked using dial gauges. The tests indicated that the realized load factor on the wing was between 9 and 10.

NRL HYDRAULIC STATIC TESTING APPARATUS

No major full-scale structural testing was performed in the next few years because of the decision to design and build a tailless glider in the Structures Laboratory[10]. This consumed both space and manpower, and it was not until completion of the fabrication of this aircraft that full scale testing resumed.

The resumption was at a much more sophisticated level, however, both because of an influx of new staff and because of a decision of the Executive Committee of the Associate Committee on Aeronautical Research[11]. The Aircraft Structures Subcommittee had recommended the installation of a large static testing frame for testing Mosquito wings for the RCAF and for tests in connection with the A.V. Roe construction plans. They suggested it be designed and constructed by A.V. Roe Canada at Malton and that it be serviced and maintained between tests by NRC. However, the Executive Committee decided in January 1947 that it would be preferable to install a "meccano" type structural steel frame, similar to the equipment at Wright Field, in the Structures Laboratory at NRC. It was considered that this would be more flexible than the conventional cathedral type frame proposed for Malton. Wings of any type, fuselages and other aircraft components could be tested by bolting together the required frame.

The apparatus was completed in early 1948[12] and the test structure itself remains in use to this day for full-scale aircraft fatigue testing. In its original form it was possible to test aircraft wings of 120 ft span and 18 ft 9 in. chord with a total upward load of 600,000 lbf plus appropriate down loads applied by hydraulic tension jacks.

The test structure consisted of a series of up to 12 gallows frames made up from commercial structural steel sections in which holes were drilled at a standard gauge and pitch and assembled by bolting in the required arrangement.

Four inverted T-section reinforced concrete beams were placed under the floor with two 15 inch steel channels back to back and one inch apart securely anchored therein. The lower flanges were welded to an 8 inch channel and the upper flanges were flush with the floor to provide an inverted T-shaped slot in the floor for anchoring the super-structure. The individual frames would be cross-braced by means of tie rods fitted with turnbuckles and end frames could be braced to suitable additional floor slots.

The main loads were applied by "Lancaster" aircraft hydraulic undercarriage reaction jacks suspended between the two channels forming the beams of the gallows frames by means of universal joints bearing in cast iron saddles clamped to the channels. Two hydraulic consoles were available which were each able to supply the hydraulic fluid at four different pressures simultaneously.

Loads were transmitted to the test article by means of "adhesive tension patches" which were at that time being used in the CSIR laboratories in Australia[13]. The tension patch consisted of a 6 inch by 20 inch patch of sponge rubber about one inch thick bonded to a steel plate provided with a pick-up point for connecting it to the loading system. The exposed surface of the rubber was bonded to the surface of the test article by means of a rubber base cement.

A group of patches was loaded by a single jack through a series of links and beams known as "whiffletrees". Structural steel channels, back to back, or rectangular section wooden members were used as beams with flat steel bars as links. A typical wing set up is shown in Figure 16.

Deflection measurements were made by means of "deflection boards". These were simply large sheets of plywood mounted in a vertical plane, with small pulleys, mounted on the top. A length of 0.01 inch diameter music wire was attached to the structure at each point where a deflection measurement was required and led over a system of pulleys, the first on the floor directly under the point of attachment, and on to the deflection board. A small weight was attached to each wire to provide tension and then any vertical displacement of the point of attachment was directly indicated by a similar displacement of the weight.

The jack loads were measured by simply reading the hydraulic fluid pressure from a Bourdon gauge since each jack was carefully calibrated prior to use.

The increase in the level of sophistication of testing at this time is also evidenced by a discussion in this report[12] of the design of the centre section anchorage to ensure that stresses in a wing for example, were accurately reproduced. Previous tests often appear to have simply used a very rigid clamping device. It is also interesting to note that the possibility of repeated load (fatigue) testing was contemplated at that time, the basic additional requirements being a larger capacity pumping system and a suitable system for cycling and controlling the loads.

MOSQUITO WINGS

The first tests performed using the new equipment were three static tests of Mosquito wings for the RCAF in the latter half of 1948[14]. The purpose of these tests was to measure the ultimate strength of each wing as an indication of the effect of service flying and storage on the plywood stressed-skin construction, and to gain experience with the new test apparatus. Two of the wings had been flown for some time in service (wings I and III) while the other (wing II) was new, having been stored for some months.

The wing was mounted in its normal horizontal position and restrained at four locations in the centre section. The rear fuselage pick-up points on the rear spar were secured to two brackets which were anchored to the test structure. The forward pick-up brackets on the upper structure of the wing just aft of the front spar were removed and two pine contour blocks were fitted to the surface in approximately the same location running from the front face of the front spar to a point 2-1/2 ft aft. The centre section was bridged by two structural steel beams bearing on the upper surface of the contour blocks and anchored to the test structure. The wing was placed in firm contact with

the contour blocks by jacking it up and inserting blocks under the front spar. These arrangements allowed the wing to rotate slightly about a fore and aft axis, but permitted negligible vertical displacement due to the rigidity of the anchorage.

Air loads were applied through 44 equally loaded tension patches on the upper surface of the wing and so positioned in the spanwise and chordwise directions to produce loading on the wing approximately equivalent to that occurring in flight under the conditions of the stressing case considered. The patches were grouped for convenience in four blocks on each semi-span, each block being loaded by one jack. Thus the air loads required a total of eight jacks.

The load from the jacks was applied to the various patches using four rows of beams made of structural steel channel except for the lowest row which was made from two pieces of rectangular pine. The rows of beams were connected with bolts by steel links of rectangular section. The links between the second and third rows of beams were twisted through 90° and the direction of the beams changed from spanwise to chordwise. The arrangement is shown in Figure 16.

Fuel inertia loads were applied to a wooden member bolted to the tank straps after removing the tanks. These loads were applied by jacks through a system of beams and links. One jack was used for each pair of outboard tanks and one for each pair of inboard tanks. Thus four jacks were required for the fuel inertia loads. Engine and undercarriage inertia loads were applied as a single load by two jacks per side through a welded steel bracket carried on the engine bearer.

A total of 16 jacks was therefore required to apply the test loads and these were supplied with hydraulic fluid from four different pressure channels. All the air load jacks were fed from one channel, the engine and undercarriage jacks from the second, while the third and fourth channels fed the outboard and inboard fuel inertia load jacks respectively.

During the test the voice of an observer describing the progress of the test was recorded on a portable Wire Recorder along with the various noises produced by local damage in the structure which were picked up by a microphone inside the wing. The tests were also recorded on film using two 16 mm cameras. Deflections were measured at 20 points along the wing using the deflection board described previously.

Failure of wings I and II occurred while holding 95% ultimate factored load as shown in Figure 17. Wing III failed while holding 85% ultimate factored load. Lacking specific information on the length of service and storage of the wings tested, no conclusion concerning deterioration could be drawn, but there was no evidence to suggest that the failures below ultimate factored load could be attributed to deterioration in service or storage.

C-102 DERWENT EMPENNAGE

The test on an A.V. Roe C-102 Derwent empennage in mid 1949[15] for A.V. Roe Canada Limited was the first full-scale test in the Structures Laboratory to use electrical resistance strain gauges. These gauges, which were manufactured in the laboratory, were used to measure the strains at various points on the structure, and to manufacture dynamometer rods which were substituted for the control rods in order that the loads on the control rods could be measured.

For the fin test, the rear fuselage section and fin structure was mounted in the testing framework with the fuselage rotated 90° from the normal flight position as in Figure 18. Tension patches bonded to the upper side were used to apply the side loads to the fin while the tailplane loads were applied through a mechanical lever arrangement. Calibrated hydraulic jacks were used to apply these test loads and lead shot bags and billets were used for loading the control surfaces. Torsional stiffness tests were performed on the fin structure and manual rudder and limit load tests on the manual and power rudders, rudder trim tab and the fin.

The tailplane was mounted parallel to the floor in an inverted position and the four forked fittings for connecting the tailplane to the lower fin were used to secure the tailplane to the test structure. Loading of the control surfaces was by means of lead shot bags and billets and hydraulic jacks were used to load the tailplane structure through tension pads. Small hydraulic jacks and tension patches were also used to load the trim tabs in the elevator limit load tests. A typical set-up is shown in Figure 19. Torsional stiffness tests were performed on the manual elevators and tailplane, and limit load tests were performed on the manual and power elevators, trim tabs and tailplane. No failures or indications of incipient failures were observed.

C-102 DERWENT MAINPLANE AND CENTRE FUSELAGE

The mainplane and centre section fuselage of the C-102 were tested in the laboratory in mid 1950 again for A.V. Roe Canada Limited[16]. The specimen consisted of the primary mainplane structure, and the fuselage centre section terminating with the transport joints. Fuselage extensions, each consisting of one bay of normal fuselage structure terminating in a short strengthened section, were fitted to the fuselage, and bolted to a rigid welded-steel ring at each end; rings were secured at two transverse pin points, by vertical links connected to the rigid steel supporting structure. Fore and aft movement of the fuselage section was limited by ties between the pins in the forward ring and a beam in the laboratory floor some distance forward of the main restraint. Figure 20 shows the test rig at different stages in the test programme.

Simulated air loads were applied to the upper surface of the wing by means of the usual calibrated hydraulic jacks through links and beams to tension patches. Concentrated loads were applied by additional hydraulic jacks through suitable linkages to fittings on the wing structure at specified locations. About 850 strain gauges were applied to the structure, and selected groups of them were recorded for the various tests.

The torsional stiffness of the starboard wing was measured and then two limit load tests were performed for a specified loading and flight condition. Several tests of the integral fuel tanks under various flight conditions were also completed. When these proved satisfactory, the specimen was tested to destruction under the same conditions as for the flight limit load test. The centre section wing collapsed at 103% ultimate factored load.

TEXAN T6 MAINPLANE

After inflight wing failures of two Texan T6 aircraft in 1951, the RCAF requested the laboratory to verify the strength of the mainplane[17]. In view of the urgency, an initial test was made using deadweights (lead shot bags), in the manner described earlier for the Harvard wings[3], in both the high and low angle of attack conditions. From this test it was concluded that the used wing which was tested satisfied the relevant statutory proof load requirements. The wing was then tested with hydraulic loading using tension patches and whiffletrees. The wing supported 120% ultimate factored load but the upper surface collapsed about 21 inches outboard of the transport joint as the load approached 125%. However, this did not duplicate the inflight failure and it was therefore suggested that the possibility of fatigue failure should be examined.

STATIC TESTS ON C-100

Over a period of seven years starting in 1950, full-scale static tests were performed on various components of the A.V. Roe C-100 aircraft, all using similar loading and measuring systems with small improvements from time to time.

The status of the National Aeronautical Establishment (NAE) with respect to the structural development of the CF-100 aircraft has never been clearly defined, but nearly all the test work was carried out on the basis of technical and financial arrangements concerted directly between A.V. Roe and NAE. The RCAF was apprised of all NAE operations, invited to witness all tests, and provided with NAE reports containing the test results.

In the case of all tests directly financed by A.V. Roe, the responsibility for the nature of the test, the magnitude and disposition of the applied loads, and the specification of the required measurements was taken by A.V. Roe. The NAE accepted responsibility solely for the methods and procedure of testing, and for the accuracy of all the applied loads and measurements.

In general, the NAE was not provided with, and was not able to obtain, sufficient relevant fundamental data to check the appropriateness of the test made, nor the validity of the test loads and distribution. The tests were therefore all conducted on the basis that the data supplied by Avro were correct, but a caveat was sent to the RCAF explaining these circumstances.

The first component tested in early 1950 was a tailplane[18]. This was the first test in which the newly developed 200-channel strain indicator was used (Figure 21). The test set-up is shown in Figure 22. A proof load test of an outer wing in the same year[19] used two slightly different loading techniques. Loads at the leading edge were applied through sheets of aluminum alloy riveted to the leading edge structure and loads at the end ribs were applied by underwing contour beams through rubber pads cemented to the lower beam. These arrangements are shown in Figure 23.

An empennage and rear fuselage strength test[20] used "compression pads" on one side of the fin in addition to tension pads on the other side to apply side loads to the fin. The compression pads were connected to the beam system by tension links passing through reinforced holes in the structure. The arrangement for this test is shown in Figure 24.

A further four outer wing tests were performed in 1950 and early 1951[21-24]. The last of these saw the introduction of load cells for measuring the loads rather than relying on pressure gauges and calibrated hydraulic jacks. This was because the nature of the required load distribution entailed hydraulics jacks loading in opposition, which would make control of the loading between increments very difficult, and since the direction of travel of the jack position could vary from increment to increment, the calibrations would not be reliable. The load cells, called "weigh-bars" were manufactured in the laboratory and consisted of a berylium-copper bar fitted with NRC bonded resistance wire strain gauges whose output was read on meters located at each jack pressure control point.

A further tailplane test was conducted in 1952[25] and compression patches were again used, this time to provide a couple at the elevator hinge. This is shown in Figure 25 which also shows a close-up of the load cell that was used.

Over the next four years four more outer wings were tested after various modifications[26-29]. Most of these tests included the effect of fuel pressurization. This was achieved by blanking the normal tank interconnections and connecting them externally in groups; pressures were increased as required by introducing fuel under pressure from a common source through separate control valves and were read on individual Bourdon gauges.

REPEATED LOAD TEST ON CF-100 WING PANELS

The first fatigue test of an aircraft structure conducted in the Structures laboratory of NAE was a manually repeated limit load test on a CF-100 Mark I outer wing panel in early 1952[30]. This wing was one of three superfluous Mark I outer wings supplied by A.V. Roe, but the tests were not requested or financed by them, but were undertaken by NAE partly as a fact-finding operation for research purposes, and partly because of some concern regarding the adoption of a 9g ultimate load factor for the aircraft. The general arrangement and linkage was similar to that described in Reference (23). However, rather than using two separately operated jacks for applying the test loads, all loading was derived from a single source of hydraulic pressure by modifying the outboard loading linkage. Load was applied manually to a maximum of 77% of the ultimate load determined in a previous test[21]; this was equivalent to a 6.5g load. A complete cycle of loading when records were not taken took place in about two minutes.

The wing began to develop serious damage after 82 cycles of load and it was expected that, in the absence of remedial action, the wing would have failed after 100 cycles in a manner similar to the mode of fracture observed in the static test. However, it was known that this mode of failure was essentially a skin instability phenomenon coupled with inadequate strength and stability of the forward ribs and rib attachments, and that the Mark I modified and Mark II wings represented an improvement in this respect. For these reasons, it was decided to effect some degree of repair and stabilization to the ribs and attachments in the hope that failure in this mode could be postponed until some more fundamental mode of failure had supervened.

The first primary structural failure apart from those occurring in the forward ribs was first noticed after 720 cycles and occurred at the main spar lower transport joint fitting. The repair scheme adopted for the ribs was successful insofar as final failure of the wing took place in a different mode from the static mode, but the final failure was not considered to have been unaffected by the rib failure. The wing withstood a total of 1543 applications of the maximum test load, and ignoring considerations of possible service loading and the behaviour of the forward rib structure, this endurance was considered to represent a satisfactory fatigue resistance for the applied load range.

The two other wings were tested at 1.5g and 3g but the results do not appear to have been reported in detail. However, in an unreferenced memorandum it was stated that while the results of the work were still being analysed, it was considered that there was no immediate danger of fatigue failures in the outer wings of in-service CF-100 aircraft.

The first automatic fatigue test of an aircraft structure carried out in the laboratory was a repeated limit load (6.67g) test on a C-100 Mark 3 outer wing for A.V. Roe (Canada) Limited in 1953[31]. The specimen was mounted on the tubular steel framework as for the static tests, and the specified shear, torque and bending moments were achieved by the use of two jack systems fed from a common hydraulic source. Load, representing wing lift, less structural and fuel inertia, was applied by four overhead hydraulic jacks, arranged in parallel, and distributed through a system of beams and links to tension patches bonded to the upper surface, aluminum alloy sheets riveted to the leading edge flap at the intermediate rib stations, and underwing contour beams at ribs 1 and 10. A secondary load, applied by a pair of smaller jacks below the wing and distributed through a system of beams and links to contour blocks fitted in the upper dive brake trough and a single tension patch on the lower surface, provided the desired modification of the primary load, particularly with respect to torque. The test arrangement is shown in Figure 26.

The automatic loading system, shown schematically in Figure 27, consisted of two deadweight pressure controllers for the upper and lower pressure limits, the movements of which actuated micro-switches and a relay, operating solenoid valves which alternately connected the jacks to a high pressure source and to a reservoir. The measurement of the jack loads was by means of strain gauged dynamometers so that corrections to the deadweights could be made as required to compensate for changes in load due to phenomena such as oil temperature variations. The rate of loading was limited to about 2 cycles per minute in order to maintain the desired accuracy of loading.

The first two cycles were applied manually in order to record strain readings before commencing automatic cycling of load. Subsequent manual loadings were carried out at 76, 1191, 1279 and 1471 cycles to investigate any differences in strain records resulting from damage. Although loud noises were heard as early as cycle 759 no damage was observed until a crack in the lower skin was observed at cycle 1218. The main spar boom failed completely after 1524 cycles and the test was halted.

MENTOR WING

In 1955 static strength tests on two specimens of the left hand outer wing of a Mentor aircraft were carried out for the Directorate of Flight Safety as part of an investigation into the cause of wing failure during a flying manoeuvre of an RCAF aircraft[32]. The wing specimen was mounted on a steel plate carried by an L-frame of steel beam construction and loaded in the normal manner to

the specified ultimate loads which were equivalent to 9g. The first wing which was removed from an aircraft after 315 flying hours when buckles were observed on the upper surface failed prematurely at 8.4g. The second wing taken from the production line, sustained the ultimate load and after a minor modification finally failed at 10.4g. Buckles which occurred in the wing during this test, and which were very similar to those which existed on the first wing, indicated that the latter had been subjected to loading in service in excess of 8g, and it was concluded that the Mentor wing structure met the design requirements of 6g limit load and 9g ultimate load. It was considered that the release of the positive uplock of the landing gear may have been an influencing factor in the service wing failure, since release of the uplock occurred during tests on the second wing at 7.7g.

REPEATED LOAD TESTS ON HARVARD OUTER WINGS

The first two-level fatigue test on a structure carried out in the laboratory was in 1957[33] when a test on a Harvard outer wing under repeated loading was performed for the Directorate of Aircraft Engineering, Royal Canadian Air Force, to investigate the cause of nose rib and stringer clip failures. A wing with at least 797 flying hours was installed as a cantilever in the testing frame and loaded by means of a single hydraulic jack through a system of whiffletrees and tension patches in the normal way.

Cycling of load was accomplished by the use of an electro-hydraulic switch gear in the hydraulic circuit, similar to that described above. A close-up of this system is shown in Figure 28. The number of cycles was recorded on a counter by the closing of a limit switch actuated by wing tip deflection.

The wing was subjected to an arbitrary program of repeated loads with the equivalent of a 5g load being applied cyclically. At intervals, the cycling was stopped while a static load equivalent to 8g was applied, this occurring after 100, 200, 400, 700 and 1100 cycles of the 5g load, and thereafter every 500 cycles to a total of 3100 cycles. Cycling was then continued at 5g to 4000 cycles and the 8g load again applied. Cycling was resumed at 6g but failure occurred after only 4 cycles. Several rivet heads had failed and the leading edge of the wing was distorted near a landing light cut-out. Removing the skin panels from the top of the surface revealed that many stringer-to-rib clips had broken, one at least showing very clear evidence of fatigue, and that the nose ribs outboard of the light cut-out had buckled.

Since there were no leading edge inspection cut-outs on this wing, there was uncertainty as to how the damage had progressed. Therefore a second wing was tested[34] with a service modification consisting of two removable panels in the lower leading edge. Repeated loading at 5g was applied as before with a statically applied load of 8g applied at 50, 100, 500 and 1000 cycles and thereafter every 500 cycles. Visual examinations were carried out in all accessible areas every 100 cycles. The mode of failure of this wing was identical to that of the previous wing. The loss of nose rib stiffness by failure of flange rivets and stringer clips led to crippling of the leading edge. From the frequent observations made during the test it was determined that the first flange rivet and first clip failure occurred after 2700 and 6100 cycles respectively. From an approximate manoeuvre load-frequency of occurrence envelope it was estimated that a load cycle of 5g might occur about once every 0.2 hours and considering only this load level, the number of load cycles of 5g from first rivet failure to final fracture (4300 cycles) could be applied during a service life of 860 hours, while from first clip failure to final failure could occur in 180 hours. It was realized that these considerations ignored the existence and cumulative effects of large numbers of smaller cycles as well as the loads in excess of 5g. However, it served to illustrate that the incidence of clip failure should be regarded as an immediate reason for withdrawal of the wing from service.

Three additional Harvard wings were fatigue tested for the Directorate of Maintenance Engineering, Royal Canadian Air Force, during 1958[35]. A modified wing with zero flying hours sustained 14,500 cycles at 5g and 31 at 8g before the test was discontinued at which time no signs of failure were apparent. A wing taken from service after field repairs was examined prior to testing by intrascope and an additional clip failure was detected and repaired. After 100 cycles at 5g the

leading edge failed at 7.6g during the application of the static 8g load. The last specimen, also taken from service after field repairs sustained 2000 cycles of 5g load, followed by 3500 cycles at +5g, -2-1/2g, 500 cycles at +6g, -3g and 3500 cycles at 6g before failure occurred. However, in this test repairs were made in the laboratory as the test progressed. It was noted that the interpretation of the laboratory tests was restricted owing to a lack of information concerning the connection between laboratory results and field effects.

REPEATED LOAD TEST ON CF-100 OUTER WING

A repeated limit load fatigue test of a CF-100 Mark 3 outer wing removed from an aircraft which had completed 1000 hours total flying time was performed for Avro Aircraft Limited in 1959[36]. This was a repeat of the test performed in 1953[31], and consisted of applying repeated limit loads (6.67g) at a frequency of about 1-1/2 cycles per minute. A crack was first observed in the lower skin forward of the main spar after 1954 cycles and final failure resulted from fracture of the main spar tension boom after 2467 cycles. This failure was across the section of the most outboard bolt of the transport joint pick-up fitting, and duplicated the final failure of the previous Mark 3 wing fatigue test. It was suggested that the appearance of a crack in the lower forward skin area could be used as a fatigue indicator. In the previous Mark 3 fatigue test more than 300 cycles were completed between the appearance of the crack (in the same area) and final failure, and in this test more than 500 cycles were completed during the same interval. It was therefore considered safe to conclude that there would be an adequate time margin between first crack and final catastrophic failure.

SABRE 5 HORIZONTAL STABILIZERS

As part of the investigation into an accident of a Sabre 5 aircraft, the laboratory conducted repeated load tests on two Sabre 5 horizontal stabilizers for the Directorate of Flight Safety, Royal Canadian Air Force, in 1961[37]. One stabilizer tested was rebuilt to new standard by Canadair Limited with no subsequent flying hours, while the second was removed from an aircraft after 746 flying hours. A special attachment was designed so that the stabilizer (less elevator) could be secured with bolts through the jack attachment in the front spar centre beam and the pivots on the rear spar centre beam. Two load conditions were used, a resultant up-load at 6.95g resulting from a manoeuvre condition with dive brakes at 50° and a resultant down load at 7.33g resulting from a balance condition. Up and down loads were applied alternately as consecutive half cycles. The up and down loads were applied through two separate hydraulic jacks, distributed through beam and link systems to tension patches over and under the stabilizer at the first four ribs. Load was measured via strain-gauged beryllium-copper dynamometers between the final beam and the loading jack and cycling was controlled automatically through an electro-hydraulic control system at about 3 cycles per minute. This was similar to that described previously but consisted of two deadweight pressure controllers, one on the "up" load jack line and one on the "down" load jack line.

Neither stabilizer showed any sign of damage after 3000 cycles at 100% limit load and so they were then tested to destruction at 125% limit load. The used stabilizer showed evidence of cracking after 1500 cycles and the front spar centre beam failed after 3374 cycles. The first crack in the rebuilt stabilizer was observed after 1355 cycles and the rear spar centre beam failed in the web after 4320 cycles.

FOUND BROTHERS FBA-2 AIRFRAME

In 1962 the laboratory conducted an extensive series of static tests on the airframe of a Found Brothers FBA-2 aircraft[38]. These were performed for the company with a view to satisfying airworthiness requirements prior to certification. Examination of the centre-section pick-up attachment prior to testing showed that its support would be unrealistic if a rigid frame section were employed and therefore the fuselage frame section was included in the wing-holding system. The entire specimen then consisted of the port and starboard wings, together with the centre-section and

tubular fuselage frame, excluding wing tip fairings, flaps and ailerons. Additional triangulated frames of light steel sections were attached to the forward and aft faces of the centre fuselage in order to apply the necessary nose and tail loads required for balance. The specimen was mounted in a freely-suspended condition in the gallows frames, being supported from the overhead beams by means of whiffletrees between hydraulic jacks on the frame and tension patches bonded to the upper wing surface. Down loads were applied to the fuselage floor by means of rods and plates leading to a fuselage whiffletree system. Nose and tail loads were applied at their relevant angles to the fuselage datum line by flat steel links connected to hydraulic jacks located on the centre-line of the aircraft in the base of the test rig. The test arrangement is shown in Figure 29.

Deflection measurements were made using deflection wires and boards, and strain measurements were made using about 300 strain gauges. Loads were measured with strain-gauged beryllium-copper dynamometers. The test loads represented the best practical compromise between the shear, bending torque and balance loads presented by the analytical design loads, and their derivation was included as an appendix to the report.

In initial tests the starboard outer panel failed at 97% ultimate design load, but after modification was found capable of carrying 100% of the ultimate design load for the symmetric loading case. In the asymmetric loading case, a bolt in the port transport joint attachment failed after 10 seconds at 100% ultimate design load.

Drop tests on the main undercarriage of the same aircraft were described in a separate report[39].

FULL SCALE FATIGUE TEST OF CF-100 AIRCRAFT

In 1963 the Directorate of Aircraft Engineering, Royal Canadian Air Force, requested the Structures Laboratory to perform a full-scale fatigue test on a CF-100 aircraft to determine whether the then accepted service life of 2800 hours could safely be exceeded[40]. An aircraft with a total flying time of 129 hours was removed from storage and made available for the test. It was a hybrid version with the extended Mark 5 tail and Mark 4 wings but was considered representative of the aircraft then in service. The complete tail section aft of the tail transport joint was removed for testing as a separate unit, and to facilitate rigging and inspection, the engines and nacelles, wing fuel cells, ailerons, flaps, various fairings and non-structural skinning, nose undercarriage and main undercarriage wheels and brakes were removed.

Wing-Fuselage Test

A flight load spectrum was determined from the only available flight data which consisted of 2783 hours of accelerometer data from eight aircraft in various squadrons. Since there were considerable variations between individual aircraft, an upper bound to the data was used except that a few exceptional records were omitted as being non-representative. No negative flight accelerations were available nor loads due to ground manoeuvring or landing. The former two were considered to have negligible effect while landing loads were accounted for by considering the maximum wing load reversal which occurred on ground when the fuel tanks were refilled.

The test load history derived from this spectrum was applied in a programmed scheme in which the load level was progressively increased and decreased in a continuous manner in repeated blocks to give the correct distribution of load level and frequency. Constant amplitude testing at the most damaging level (from a Miner analysis) was rejected as being only able to show the mode of failure pertinent to the selected load level. In addition, the interpretation of the test would also be dependent on the validity of the cumulative damage hypothesis used. Random load application according to the load and frequency distribution was rejected both because of its complexity at that time and because of a concern over interpretation of a test result which might be influenced by the particular random sequence used in the test.

Since the shortest unfactored predicted life for the CF-100 was 4470 hours, the block length was made 50 hours and the lower and upper truncation levels set more or less arbitrarily at 2g and 6g respectively. The upper load level, which corresponded to 82% of design limit load was applied once in each 50 hour block. Down loads equivalent to the maximum fuel weight were applied in groups of 25 cycles at the end of each 50 hour block. It was recognized that the influence of these down loads would be more severe if they were uniformly distributed throughout the loading rather than lumped together, but the application of these loads required a test configuration change, since the weight of the aircraft had to be transferred from the wings to the landing gear. Distribution of these loads would therefore have been uneconomical in terms of test duration. Air loads and inertia relief due to wing and fuel weight were calculated for the maximum load factor of 6.0 and the ground loads calculated for the static case at maximum gross take-off weight with a full fuel load in each wing and tip tank.

The prime consideration in applying the test loads was to ensure that the vertical shear and spanwise bending moment distributions on the wing were correct. The wing-fuselage combination was suspended in the structural test frame, effectively from a point on each wing corresponding to the centre of lift, through a whiffletree arrangement connected to a number of loading pads. These were secured to the upper skin surface at the intersection of the ribs and the front and main spars, and were lightly cemented to the skin and held in place by bolts passing through the wing. Good agreement was obtained between theoretical and applied shear and bending moments but the torque about the main spar was only 60% of the required value since this would have required a marked deviation from the theoretical requirements for the fuselage load distribution to maintain equilibrium.

Distributed loads were applied along the fuselage through a series of beams and links by four hydraulic jacks, one applying the balance of the required nose inertia, one the engine inertia, one the balance of the aft section and tail section inertia and the fourth balancing the test weight of the centre section and any small increment necessary to maintain equilibrium. The inertia loads of the centre section were considered balanced by the lift attributed to this section. Attachment to the fuselage was through jacking pads, riveted straps, engine bearers, special attachments riveted to the fuselage formers and the tail cone attachments. A dummy strut secured between the nose undercarriage pivot centre and the ground, in addition to providing nose gear reaction in the landing case, maintained pitch control by reacting any small imbalance in the system as the wings deflected under load.

The static ground case was achieved by raising the complete aircraft by means of hydraulic jacks pushing on the main undercarriage legs so that the wing was relieved of all up-loads. Fuselage down loads were applied by the same system as for the flight loads and wing down loads were then applied by two jacks connected to a dead weight through underwing whiffletree systems and pads attached to the lower skin surfaces similar to the upper ones. The final link between the jack and the deadweight was designed with free movement to ensure that the weight would not be picked up during application of flight loads. Figures 30 and 31 show front and side elevations respectively of the basic test frame and loading systems. Figure 32 is a photograph of the complete test arrangement.

The load control system consisted of four basic units, a function generator, regulated d.c. reference voltage and discriminator, and programmable frequency control and amplitude control potentiometers. Programs were etched on conductive paper, the etched line dividing the surface into two electrically isolated areas. The paper was attached to a rotating drum and an electrostatic probe connected to a potentiometer followed the line to give an output proportional to the position of the line. The function generator generated a sine wave function in which the base frequency was varied by the control potentiometer so that the small cycles could be applied faster than the large ones. This output was then modified by a bias voltage to shift the mean to the desired level and then further modified by the amplitude control potentiometers which resulted in an output amplitude corresponding to the desired load history. This output was then divided by proportioning attenuators so that the signal being fed into each of the three servo-controllers was in direct proportion to the load requirements of each system. The load applied to the centre section lift system was maintained at a constant value throughout the test by a set-point adjustment of the corresponding servo control unit. The frequency was such that a complete 50 hour load block consisting of 454 cycles was applied in 2-1/4 hours.

The first block of automatic spectrum loading was completed on May 28, 1964 and continued until the 86th block (4300 equivalent flight hours). During this block, a 2-1/2 inch long crack was detected in a main gear door beam bracket (a secondary structure) and it was established that similar cracks had occurred on several service aircraft. The area was repaired but the repair failed after a few blocks and after further unsuccessful repairs the rate of crack propagation eventually decreased so that by the end of the test it was virtually dormant. During the remainder of the test up to block 400 (the original target of 20,000 equivalent flight hours) a number of minor failures occurred but only one was considered of any consequence. This was a failure of some bolts attaching the lower wing skin to the spar near the port outer wing transport joint, and those were then replaced every 100 equivalent flight hours (one major inspection period).

During block 403 small cracks were observed in the steel fingers of the lower starboard wing transport joint fittings and were subsequently observed in the other wing transport joint fittings. The test was continued without any attempt at repair since any repair would be a very major undertaking and it had already been demonstrated that the structure had an adequate life for the known RCAF requirement. Final catastrophic failure of the port lower transport joint occurred at the maximum load during block 483 (24,150 equivalent flight hours).

Using a safety factor of 5.0, it was concluded that under the specified loading conditions the safe fatigue life of the CF-100 aircraft was approximately 4830 hours, but it was recommended that a continuing program of load monitoring be carried out on service aircraft in order to validate the load spectrum used in the test programme.

Horizontal Stabilizer Test

For a given manoeuvre the pilot-induced loads on the stabilizer can vary by a substantial amount depending upon the rate of application of control angle. Since there was no statistical data on this distribution it was decided simply to use the values of maximum total symmetrical load determined in a previous study of the symmetric tail loads of this aircraft[41], divided by the acceleration increment of the manoeuvre. This gave an average value of tail load per g of 1000 lbf. For each manoeuvre initiated by a negative tail load it was conservatively assumed that there would be an equal and opposite tail load to terminate the manoeuvre. With the tailplane mounted on the fin, any rolling moments applied to it must be carried by the fin. Therefore, since in addition there was some doubt as to the fatigue resistance of the tail cone transport joints with respect to asymmetric loads, the tailplane was subjected to both symmetric and asymmetric loads. It was decided that each symmetric load application be followed by an equal asymmetric load giving a loading sequence — symmetric up, asymmetric port down, symmetric down, and asymmetric starboard down as shown in Figure 33. There was no direct correspondence between the tailplane flight and test loads other than the general form of the load spectrum described previously. The emphasis on the tail was therefore to determine the mode of failure rather than the safe life.

The complete tail section less the top portion of the fin, the rudder, tail cone and elevators was attached to a tubular steel test frame at the four transport joint fittings with the fin vertical. A constant down load equal to the required maximum (5000 lbf) was maintained by a deadweight connected to the lower loading system. The loading jacks, secured in universal mountings within the gallows frames, were then used to apply a net up-load through the upper loading system capable of varying between zero and twice the deadweight giving an effective net load on the stabilizer varying between -5000 lbf (down) and +5000 lbf (up). Corrections were, of course, made for the tare weights of the stabilizer and loading systems. The loading systems consisted of an upper and lower whiffle tree connected to pads lightly connected to the skin surfaces and held in place by bolts passing through the stabilizer. Figures 34 and 35 show rear and side elevations respectively of the basic test frame and loading systems. Figure 36 is a photograph of the complete test arrangement.

The load control system was similar to that for the wing-fuselage test except that the frequency was kept constant and two additional programmable amplitude controllers were used, one for each side. The final output to each servo control unit was proportional to the product of the primary and secondary amplitude signals as shown in Figure 37.

During the greater part of the fatigue test, only minor damage occurred, usually rivet failures. After completion of 300 blocks (12,000 equivalent flight hours) repairs were made since these would normally be made on service aircraft during overhaul. Similar repairs were made after 16,300 equivalent flight hours and a very thorough inspection carried out at 20,230 equivalent flight hours found no major defects. Since there were some doubts as to the validity of the test loads and the test had to continue until a major failure had occurred, the symmetric loads were increased arbitrarily by 50%. After an additional 500 blocks (a total of 40,230 equivalent flight hours) there was still only minor damage. Testing was then continued at constant amplitude at the highest load the test rig was capable of (±7.5g). After 21,800 of these cycles, a 2 inch long crack was observed in the starboard angle of a former. After a total of 42,700 constant amplitude cycles, the crack was still propagating slowly and had been present for a sufficient length of time to ensure that it could be detected in service prior to any major failure. The test was therefore discontinued.

FULL SCALE FATIGUE TEST OF CT-114 TUTOR WING

In 1968 the Directorate of Aeronautical Engineering and Simulators (DAES), Canadian Forces Headquarters, requested the assistance of the Structures Laboratory in resolving a premature cracking problem in the Canadair CL-41 Tutor aircraft[42]. The aircraft had been designed and tested to meet the requirements of U.S. military specification MIL-A-8866[43] and had been tested to an equivalent service life of 40,000 hours with no significant failures. However, 63 cracks had been reported in the upper spar caps of the 48 aircraft in service, the first cracks being found in service aircraft at lives ranging from 900 to 1700 flying hours. Similar cracks in the fatigue test airframe were estimated (then) to have developed shortly before the completion of the 40,000 hour test. The location of the cracks was also disturbing, since the loading on the upper element of the main spar was presumably primarily compressive, whereas fatigue is usually associated with tensile stresses.

After reviewing the original fatigue test, only two possible question areas arose. One was that Canadair used a low-high load application sequence rather than the then currently accepted lo-hi-lo sequence, and the second was that 8.8g loads were included every 300 hours although the aircraft would not encounter them that often.

It was therefore decided to measure the strains near the area of interest on an instrumented aircraft and perform a further fatigue test on the wing with a more representative spectrum. In addition, the upper spar caps were studied to check the metallurgical condition and residual stress levels.

First Phase Fatigue Test — Trainer Role

The derived spectrum for the test[44] was based on counting accelerometer data from the more severe of two training bases. The loads were applied in a programmed sequence of progressively increasing and then decreasing loads of the same general form as for the CF-100 test, with each positive load cycle being followed by the negative equivalent that had the same frequency of occurrence. Since it was considered that a minimum of 30 load blocks should be applied before failure to avoid block sizing effects, and cracks had been observed as low as at 800 hours, a block size of 10 hours was chosen. Since during a typical 10 hour block only peak loads corresponding to 5.4 and -0.55g were encountered, the missing more severe loads were applied manually at the end of a group of five 10 hour blocks, so that all loads that were present in the load spectrum were applied during the course of 1000 equivalent test hours. The truncation level was set at a level that was exceeded 100 times per flight hour, which resulted in 1000 cycles per 10 equivalent flight hours compared with the 145 applied by the manufacturer.

The test specimen consisted of the forward and centre fuselage sections less all major removable items, and both wings complete with main landing gear but less ailerons and flaps. The fatigue set-up, shown in Figures 38 to 40, was based on the original Canadair test, but all loads other than those due to normal acceleration were omitted. In addition some whiffletrees were extended so that only four actuators were required compared to the 43 of the original test. The complete test

airframe was suspended from a point on each wing corresponding to the main centre of lift and pivoted about the nose gear retraction axis. Down loads were applied to each wing through a beam and link system as shown in Figure 38, and these, together with inertia loads, were balanced by an up-load distributed along the fuselage centre line through a second beam and link system shown in Figure 39. These applied loads remained constant throughout the test and taken in combination gave the required net shear and bending moment at each wing station for the -2.0g flight case. The geometry of the whiffletree arrangement resulted in a good match between the theoretical and applied wing shears and moments. Similarly the applied and theoretical torques along the main spar, while not so exact, also produced an acceptable simulation of the theoretical forces[45].

For all cases other than -2.0g, down loads were applied along the fuselage centre line, again through a beam and link system, reacted at the wing suspension points and appropriately distributed along the span and chord by the over-wing beam and link system.

The total applied fuselage down load was varied between zero (for the -2g case) to a maximum calculated to produce the required net shear, bending and torque at each wing station and the wing/fuselage joint for the 7.8g maximum positive flight case. Thus the various forces and moments were assumed to vary linearly with aircraft acceleration.

The analogue command input to the servo-hydraulic system was generated by an electro-mechanical function generator. The basic sinusoidal voltage derived from a conventional low frequency oscillator was fed in turn to each of a group of ten counters. With each counter were associated an attenuator to reduce the oscillator output to the required amplitude, and summing circuitry to add a d.c. or steady load component. As each counter ran down to its preset value, it tripped a stepping switch that in turn diverted the output to the next counter. The stepping switch had a total of 20 positions and permitted each of the counters to be used more than once in a given sequence if needed. For the purposes of this test, only nine of the ten possible counters were employed, and eight of those were used twice to produce a conventional low-high-low block loading sequence as shown in Figure 41.

Frequency of loading was initially low but was gradually increased to a maximum of one 10 hour block in 71 minutes. This speed was limited by the servovalve capacity at the top and bottom fuselage actuators and by servo stability considerations.

Cracks were first observed in the upper spar cap at 850 equivalent flight hours, and after 2000 hours, the initial target, the wings were removed for a detailed examination of the cracks. They were found to be 1.9 and 3.6 mm in length on the port and starboard spar caps respectively, and the test had met its original objective of reproducing typical service cracks.

Parallel work on the metallurgy and residual stresses of the spar cap[46,47], coupled with a knowledge of the operating stresses[48] and strain gauge observations on the test rig had meanwhile suggested the cause of the cracking[49]. It was thought to be caused by compressive yielding resulting from the addition of normal compressive stresses and the compressive residual stress from manufacturing enhanced by the stress concentration in the vicinity of the spar cap flange root. This produced a residual tensile stress when the compressive stress was removed as the wing structure was unloaded from its high g state.

Since the residual stress caused by the manufacturing process diminished with depth, it was suggested that there would be a depth at which crack growth would cease. This depth would depend upon the balance between diminishing crack growth due to the crack propagating into regions of lower residual stress and the accelerating rate caused by the increasing stress level due to the presence of the crack.

Therefore the crack in the port wing was simply blended out in the then current service fashion and the testing continued. The starboard wing, however, was modified to test a specific repair scheme.

The wings were removed again for inspection at 4000, 4600 (port only), 6000 and 8000 hours and at this last inspection cracks were observed to have reappeared on the port wing upper spar cap. They were very small, however, and no modifications were performed.

Phase one testing was halted in April 1971 at 30,000 equivalent flight hours (in the training role). A post test inspection revealed no significant damage.

Second Phase Fatigue Test — Snowbird Role

In developing the original test spectrum it had been assumed that the aerobatic role that the Tutor was used for at that time, the Golden Centennaires, would not continue. However, although the Centennaires disappeared they were succeeded by another aerobatic display team, the Snowbirds. The spectrum for this role was similar to that for the trainer role in all respects except the frequency of high g manoeuvres and fatigue damage calculations showed that the Snowbird role was 12 times more damaging than the trainer role.

Since the fatigue life of the aircraft in this role could be consumed in the foreseeable future, it was decided in November 1975 to resume the Tutor fatigue test[42]. Because of restricted manpower, it was necessary to shorten the test and rather than delete the very numerous low loads from the test it was decided to alter the loads spectrum to correspond to the Snowbird role. It was recognized that the resulting test life would be a combination of the two roles which might pose problems of interpretation. It was anticipated, however, that the test would disclose the mode of failure and so provide guidance for scheduling inspections and other precautions to ensure the flight safety of the aircraft.

The test spectrum was based on counting accelerometer data that had been accumulated from seven Snowbirds and was defined in twenty steps, with a truncation level of 2.15g compared with 1.93g of the first phase. However, in order to speed up the test, after 7000 equivalent Snowbird hours, the last two levels were omitted resulting in a truncation level of 2.77g. Since more than 2.25×10^6 cycles of 2.5g or less had previously been applied in the first phase test, it was assumed that any fretting damage due to these loads had already occurred and so this truncation was not unreasonable. Although the Snowbird role involves several manoeuvres at -3g, negative loads were truncated at -2g because of rig limitations. This was calculated to introduce an error of about 2.3% on the fatigue life[50].

The fatigue test loading system used was the same as for the trainer role, but the method of generating the loads was quite different. The laboratory had acquired a PDP8e computer with 4K of memory since the completion of the trainer test and computer programs that had been developed permitted a much greater versatility in the generation of load time histories. Since a random sequence was considered to be more representative of actual service than any ordered sequence, the flight loads were therefore randomly generated on a draw without replacement basis. The loads table was set up so that one complete block represented 100 equivalent flying hours, and each block was randomly generated. Thus the sequence of loads in each block was different.

Because it is very difficult to determine crack propagation rates from post-fracture analyses of fractures produced during the course of a random sequence fatigue test, it was also decided to use a "marker block" in the load time history. This consisted of 64 constant amplitude cycles that were applied as the concluding segment of each 100 equivalent flight hour block. The purpose of this marker block was, after crack initiation, to provide an identifiable region of crack growth (marker band) within every 100 hour load block. By observing the spacing of successive marker bands it would then be possible to determine crack growth rates. It is important to note that this marker sequence did not alter the total damage content of the applied loads environment of the test structure, since these 64 cycles of load were extracted from the original load time history. It was recognized, however, that some of the sequential effects of these loads would be lost, but by choosing an intermediate load level for the marker (2.221g ± 3.031g compared to a maximum of 3.116g ± 5.116g) this was thought to be minimal and far outweighed by the potential benefits.

Testing was resumed in early 1976 and at slightly in excess of 2100 Snowbird hours, a crack was found in the tang of the lower main spar cap at the port wing root. This crack eventually propagated until the lower main spar cap was completely severed at 10,200 Snowbird hours. Post-failure fractographic studies[51] disclosed that this crack had been present since the late stages of the phase one test, but was not discovered at that time. Once discovered, the crack was readily observable visually, by ultrasonics and by dye penetrant techniques.

At 4500 Snowbird hours, a similar crack was found on the starboard wing. Because of its later initiation and the general slow initial crack propagation rates, this crack did not extend to catastrophic dimensions by the end of the test.

The conclusion of the test came when the port wing main spar separated at 10,200 Snowbird hours, and the rear spar failed simultaneously due to partial fatigue damage as well as to the sudden overload when the main spar support was lost. This failure, which occurred at the spar fuselage joint, was not detected during the course of the test.

The main spar top flange cracks that generated the initial fatigue life concern did not propagate further during this test.

FULL SCALE FATIGUE TEST OF CS2F TRACKER WING

In the mid seventies, a review by the Canadian Forces (CF) indicated that the Grumman CS2F Tracker aircraft could continue in its maritime reconnaissance role until the mid nineteen nineties. This necessitated an audit of the aircraft structure to ensure that the airframe would remain airworthy for this period. A previous Tracker fatigue test performed in Japan for the United States Navy was not considered relevant to the proposed Canadian operation because of differences in load spectra and difficulties with interpretation of the Japanese test. Therefore in June 1976 the Department of National Defence initiated the Tracker fatigue life extension program which involved (a) a full scale fatigue test of the Tracker wing, (b) a Grumman Aerospace Company review of the Tracker fatigue analysis using CF mission and spectra data and (c) a limited damage tolerance assessment of the wing-to-fuselage attachment fittings, a lower wing lock fitting and a lower wing stringer. NAE was requested to perform the full scale fatigue test[52].

A large amount of flight data for the Tracker was available from counting accelerometers. Although it was representative of current usage it contained an unusually large number of 4g exceedances for a 3g limit design load aircraft. It was suspected that these high counts were either put on inadvertently during servicing of the accelerometer systems or occurred during rocket firing exercises that were carried out several times a year. The latter reason was not expected, however, and a later load measurement program during rocket runs[53] confirmed this. The decision was therefore made to statistically edit the data to remove some of the high g counts. The data were assumed to have a log-normal distribution of counts about a particular 'g' level and only those periods of data for which all the data fell within a band on either side of the mean number of counts/hour were accepted. Several different band widths were tried and the final rejection criterion used was to remove all periods of data which contained counts which fell outside 2.17 standard deviations from the mean. Using this technique approximately 10,000 hours of the 16,000 hours of data were retained[54]. Since there were few measured ground load accelerations for the Canadian Tracker aircraft, the ground loads spectrum was developed from the ESDU Data Item 75008.

The final flight load spectrum was split into a symmetric gust load spectrum and a non-symmetric manoeuvre load spectrum and these two spectra were then discretized into intervals of 'g' containing inegral numbers of cycles in each block of 100 flights. This resulted in 20 steps for the manoeuvre spectrum and 15 for the gust spectrum. A lower truncation level of 1.25g was used for the positive spectrum because a preliminary fatigue life estimate indicated that over 98% of the fatigue damage was caused by positive accelerations above this level. An upper truncation level of 3.84g was chosen using Maxwell's guideline[55] that the maximum test load should be that which is seen approximately ten times in the life of the aircraft.

The test loads were applied in blocks of 100 flights each representing 3.6 hours of service including taxi time, with a nominal testing flight time of 3.5 minutes. Each flight consisted of one pre-flight ground load, 49 manoeuvre cycles, two gust cycles and five post-flight ground cycles. To compose the first 99 flights, the 51 manoeuvre and gust cycles were randomly selected on a draw-without-replacement basis from a load table representing 360 hours of the derived spectrum. A similar random selection of six ground cycles completed the flight. The 100th flight was a marker block consisting of 51 constant amplitude cycles of between 1.878g and 0.912g. This marker flight was applied to mark any developing fracture surfaces with recognizable bands for use during post-test fractographic analysis[56]. These marker cycles were not an addition to the load spectrum but were 'saved' from the total applied load table.

The test specimen consisted of a complete aircraft less the nose section and tail section. Most electrical and electronic equipment and wiring was removed along with non-structural doors and panels, flaps, ailerons, slats, wing tip fairings and pylons, wing fold actuators, landing gear and engines. It was suspended from a point over each wing in the standard NAE test frame via whiffle-trees as shown in Figure 42. Previous NAE full-scale tests used steel whiffletree loading systems. For this test, 6351-T6 aluminum channel was selected to save both weight and cost. The maximum operating bending stress of 10,000 lbf/in^2 used for designing previous steel whiffletrees was maintained and the design procedure was experimentally verified[57]. Chordwise load distribution was obtained by using four inch wide contour boards running from the 15% to 57% chord at each rib of the outer wing. These contour boards were bonded to rubber patches and then bonded to the wing surface; no through holes were used. The peak tensile stress between the contour board and the wing surface was limited to 20 lbf/in^2 [58].

The Canadian Forces Directorate of Maritime Air provided the mission usage data and a hypothetical average flight condition was defined. The net wing shears, bending moments and torques for this condition were then calculated[59]. Because the fatigue critical area was expected to be just outboard of the wing fold, an effort was made to closely match the bending moments, shears and torques in this region[60]. This was achieved by six loading systems shown in Figures 43 to 46. The under fuselage system, Figure 43, simulates the fuselage inertia loads and matches the fuselage bending moments. The over fuselage system, Figure 44, simulates the centre wing inertia and air loads, and the ground loads for the ground spectrum. The under nacelle system, Figures 44 and 45, simulates the engine inertia and ground loads. The underwing system, Figure 44, simulates the down loads on the wing and in conjunction with the overwing system applies the best fit shear, torque and bending moment distribution for the whole flight spectrum. The overwing system, Figures 44 and 46, reacts all the vertical loads applied to the aircraft and the nose pivot point reacts any fore and aft loads applied to the structure.

Since only symmetrical loading cases were considered, only four command channels were required for the six servo-hydraulic actuators. The load level was randomly generated via the PDP8e computer used for the Tutor test, and then the command signals, which were all a simple linear function of the load level, were calculated for each channel. All of the actuators were fitted with load limiters (pressure relief valves) which protected the test article from being damaged by overload. They were also fitted with abort modules which allowed the actuators to be locked in position in the event of a problem with the system. Pressure in the actuators could then be released manually by valves in the abort modules.

Testing commenced on May 3, 1979 after an uneventful start-up. Sets of strain, load and deflection data were recorded incrementally on a routine basis every 5000 flight hours and before removal and after reinstallation of the outer wings which was generally every 15,000 hours. Numerous cracks were observed and monitored during the course of the test at locations such as the nacelle firewall skin, spanwise stringers, door frame and fuselage frames. One area of interest was the search-light cut-out of the starboard wing. At flight 4000 a crack was discovered in a lower-skin doubler strap at the cut-out, which was initially thought to be insignificant. However, at flight 4332 (15,595 hours) a 'bang' redirected attention to this area, and an additional crack was found in the vertical leg of the front spar cap angle near the cut-out. The test was restarted and the crack grew rapidly to a captive nut attachment rivet hole. At this stage, it was decided, in liaison with DND and Grumman,

to suspend testing until a repair had been made. The elapsed time from suspension of the test to the finished installation of a Grumman developed repair scheme was approximately five months. Subsequent inspections of this same area on the DND fleet of Tracker aircraft discovered three similar cracks, except that they had not progressed into the spar cap angle. The same repair scheme was applied to these aircraft.

The most prolific type of cracking observed during the test was that initiating in the lower wing skin where an internal doubler was spotwelded to the new skin panels. The first crack was found during the 60,000 hour inspection, and once aware of the type of crack, inspection of other spot welds resulted in the discovery of an additional 28 crack locations. Merging of several of these spot weld cracks on the right hand wing finally led to failure of the wing on application of the highest load during flight 32541 (117,144 simulated flying hours) on October 29, 1982. It was concluded[61] that, using a safety factor of 4, the "safe-life" of the wing and its attachment to the fuselage for operational aircraft under the current flying conditions was 29,000 hours. In addition, the growth of the lower skin cracks which led to final failure was slow enough to ensure a high probability of detection well before any possibility of catastrophic failure.

CONCLUSIONS

Thirty-eight full scale tests of large aircraft structures have been performed in the Structures Laboratory of NAE in the last 43 years. Table 1 provides a summary of these tests including the agency for which the tests were performed, and it can be seen that over 90% of them have been performed for or in support of the Department of National Defence and its predecessors.

In the early years the laboratory performed many original tests for DND that could not be performed elsewhere in Canada and played a major role in performing structural tests on the CF-100. In later years, as the Canadian aircraft industry took on most of its own testing, the laboratory has acted in a more supportive role, providing DND with an independent source of expertise and advice. In addition, it has been able to provide alternative facilities for full scale fatigue testing of large structures, which are not always available in industry on a continual basis.

The complexity of full scale aircraft structural testing has increased dramatically, however, over the past 40 years. Whereas the initial tests involved a few people for a few days, the last test on the Tracker took about 20 man years to set up and ran in the laboratory for over 3-1/2 years. Similarly, whereas the first tests were performed with rulers and lead shot bags, the last test involved computer controlled servo hydraulic actuators. It is therefore clear that NAE must be prepared to make major committments both in equipment and manpower if the laboratory is to maintain its presence in this field.

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TABLE 1

LARGE SCALE TESTS AT NAE/NRC

YEAR	A/C TYPE	TESTED FOR	TEST DETAIL
1941	Anson	DND	Fuselage — Static Load
1942	Tiger Moth	DND	Upper Mainplane — Static Load
1942	Harvard II	DND	Mainplane — Static Load
1942	Hurricane	British Purchasing Comm.	Wing Panel — Static Load
1942	Harvard	DND	Rear Fuselage & Tail Surface — Static Load
1943	Anson	DND	Mainplane — Static Load
1943	Anson	DND	Tailplane — Static Load
1944	Cornell	DND	Complete A/C — Drop Test
1944	Cornell	DND	Wing — Static Load
1948	Mosquito	DND	Wings — Static Load
1949	C-102 Derwent	A.V. Roe	Empennage — Static Load
1950	C-100	A.V. Roe/DND	Tailplane — Static Load
1950	C-100	A.V. Roe/DND	Outerwing — Static Load
1950	C-100	A.V. Roe/DND	Empennage & Rear Fuselage — Static Load
1950	C-100	A.V. Roe/DND	Outerwing — Static Load
1950	C-100	A.V. Roe/DND	Outerwing — Static Load
1950	C-102 Derwent	A.V. Roe	Mainplane and Centre Section Fuselage — Static Load
1951	C-100	A.V. Roe/DND	Outerwing — Static Load
1951	Texan T6	DND	Outerwing — Static Load
1952	C-100	A.V. Roe/DND	Outerwing — Static Load
1952	CF-100	Internal	Outerwing — Repeated Limit Load
1952	C-100	A.V. Roe/DND	Tailplane — Static Load
1953	C-100	A.V. Roe/DND	Outerwing — Static Load
1953	C-100	A.V. Roe/DND	Outerwing — Repeated Limit Load
1955	C-100	A.V. Roe/DND	Outerwing — Static Load
1957	C-100	A.V. Roe/DND	Outerwing — Static Load
1957	Harvard	DND	Outerwing — Repeated Two Level Loading
1958	CF-100	A.V. Roe/DND	Outerwing — Static Load
1958	Harvard	DND	Outerwing — Repeated Two Level Loading
1959	Harvard	DND	Outerwing — Repeated Multilevel Loading
1959	CF-100	A.V. Roe/DND	Outerwing — Repeated Limit Loading
1961	Sabre 5	DND	Horizontal Stabilizer — Repeated Two Level Loading
1963	FBA-2	Found Brothers	Wing/Fuselage Frame — Static Load
1963	FBA-2	Found Brothers	Main Undercarriage — Drop Test
1965	CF-100	DND	Wing/Fuselage & Tail — lo-hi-lo Block Loading
1968	CL-41 Tutor	DND	Wing/Fuselage Attachment — lo-hi-lo Block Loading
1975	CL-41 Tutor	DND	Wing/Fuselage Attachment — Random Loading
1977	CS2F Tracker	DND	Wing/Fuselage Attachment — Random Loading

FIG. 1: PROOF LOAD TEST ON AN 800-HOUR TIGER MOTH WING

FIG. 2: PROOF LOAD TEST ON AN 800-HOUR TIGER MOTH WING

FIG. 3: HARVARD II MAINPLANE AT 90% SPECIFIED ULTIMATE LOAD

FIG. 4: HARVARD II MAINPLANE SHOWING 'NORMAL' WRINKLING AT 90% ULTIMATE SPECIFIED LOAD

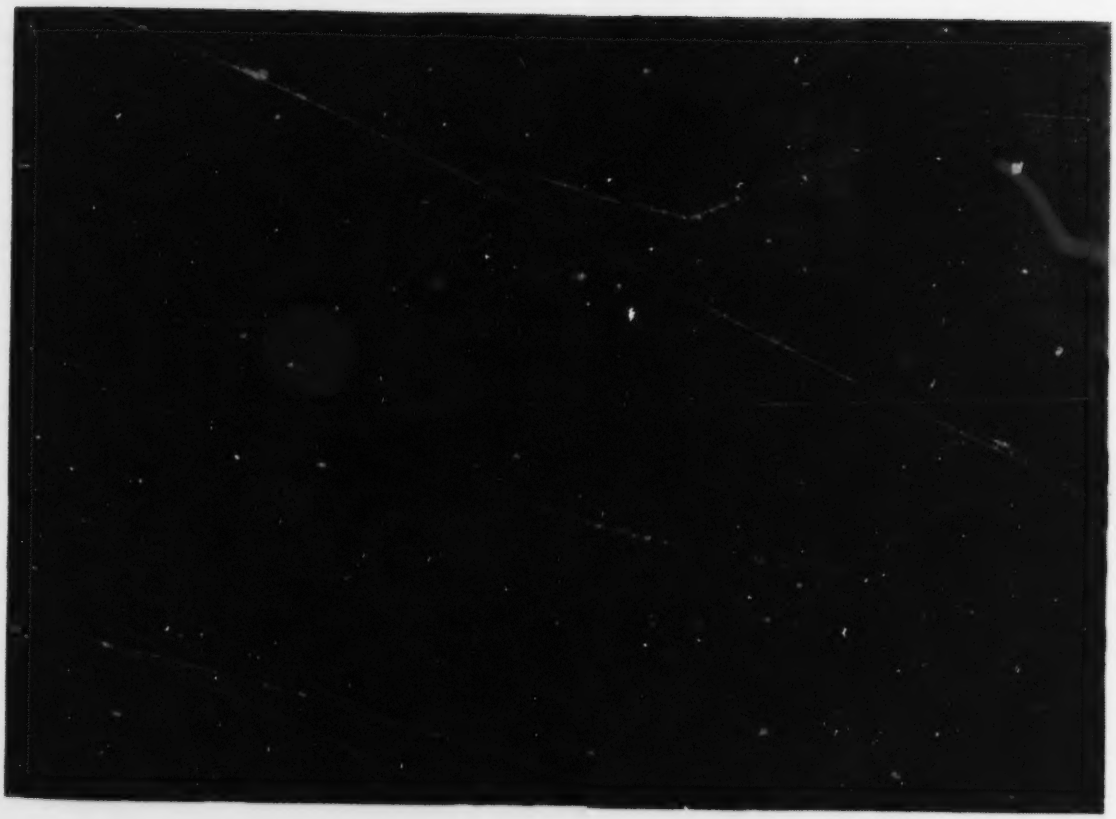

FIG. 5: MOULDED WOOD HURRICANE WING PANEL

FIG. 6: MOULDED PLYWOOD HARVARD FUSELAGE UNDER ULTIMATE DESIGN LOAD FOR A SPECIAL TAILWHEEL CONDITION AFTER REPAIR

FIG. 7: HORIZONTALLY MOUNTED TEST OF MOULDED PLYWOOD HARVARD FUSELAGE

FIG. 8: FAILURE OF MOULDED PLYWOOD HARVARD FUSELAGE UNDER A SPECIAL TAILWHEEL CONDITION

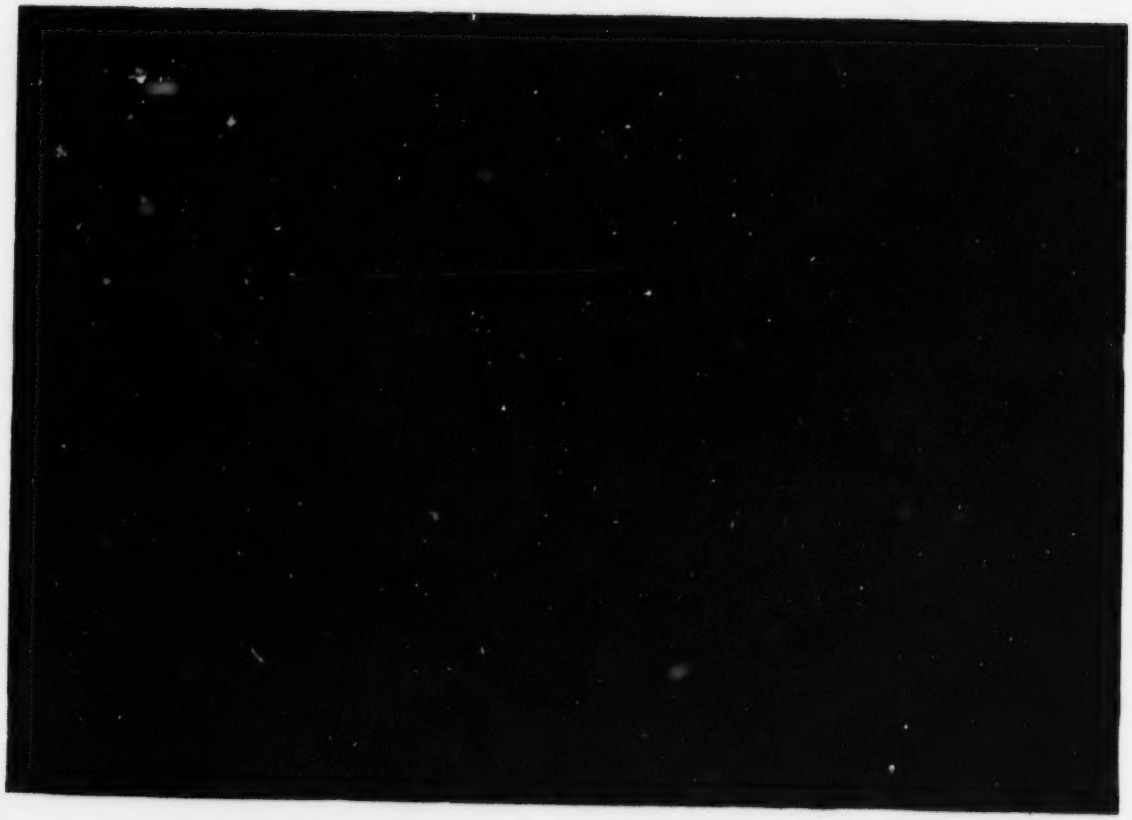

FIG. 9: STATIC TEST OF MOULDED PLYWOOD HARVARD STABILIZER

FIG. 10: STATIC TEST OF MOULDED PLYWOOD HARVARD ELEVATOR

FIG. 11: TEST OF ANSON I MAINPLANE SHOWING SLATS USED TO DISTRIBUTE SHOT BAG WEIGHTS

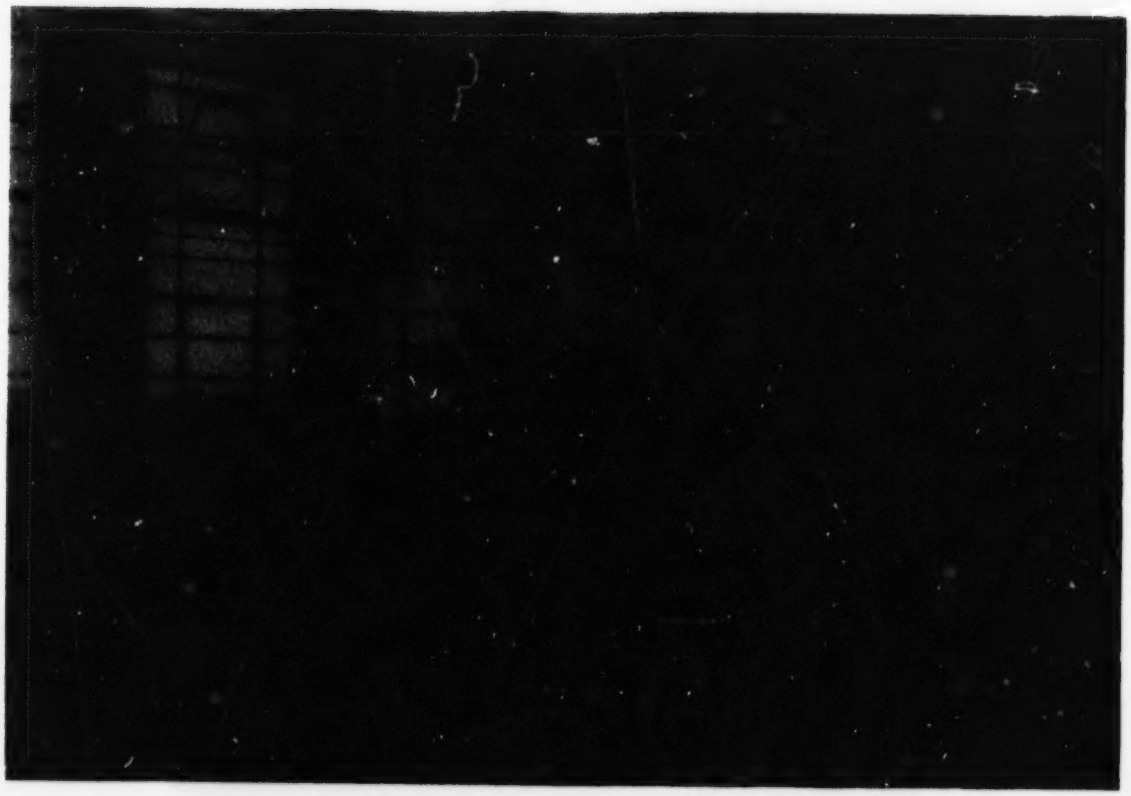

FIG. 12: ANSON MAINPLANE UNDER LOAD SHOWING WHIFFLETREE FOR APPLYING ENGINE AND UNDERCARRIAGE LOADS

FIG. 13: TEST SET-UP FOR ANSON TAILPLANE

FIG. 14: DROP TEST OF A CORNELL AIRCRAFT

FIG. 15: TEST SET-UP FOR CORNELL WING

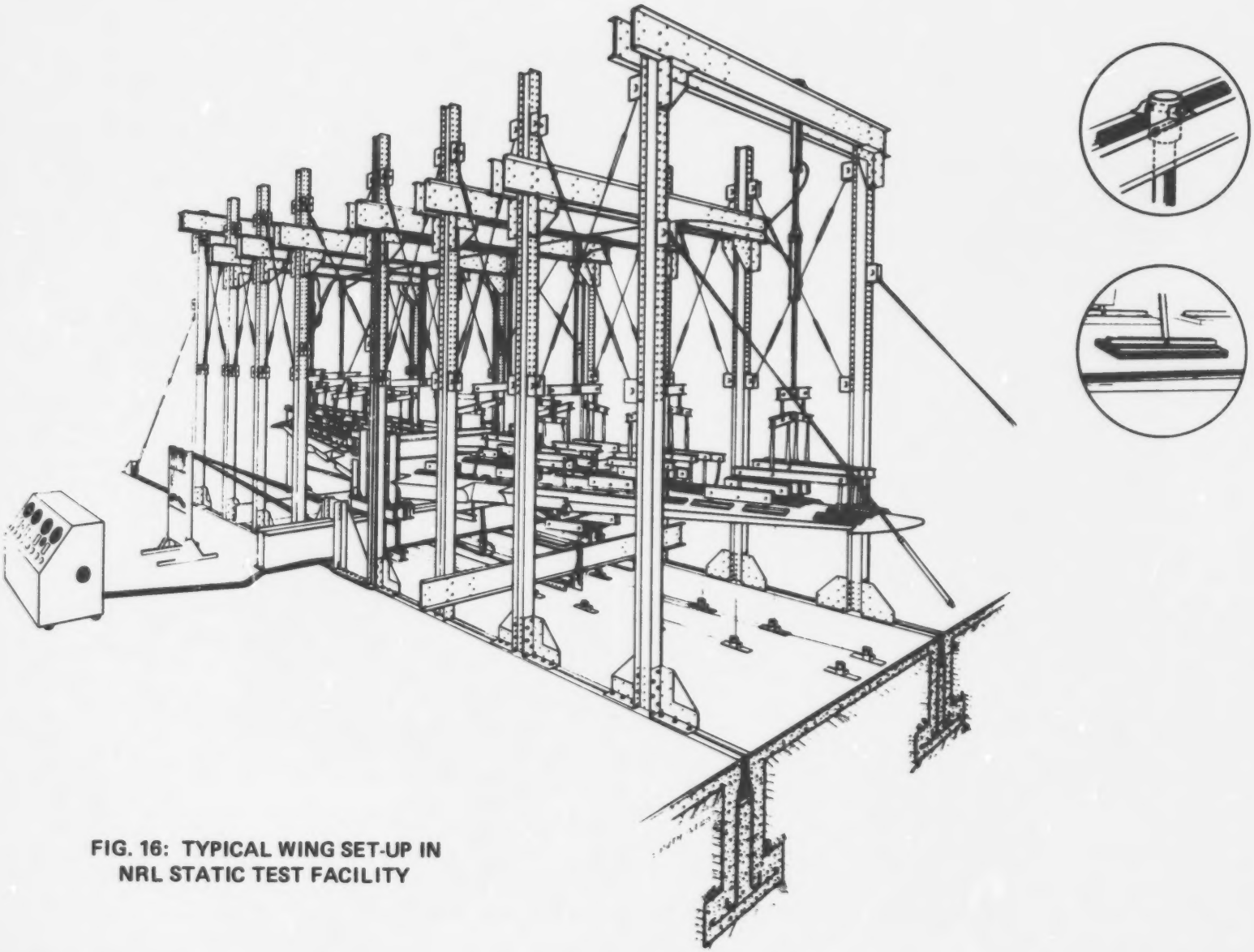

FIG. 16: TYPICAL WING SET-UP IN
NRL STATIC TEST FACILITY

FIG. 17: FAILURE OF A USED MOSQUITO WING AT 95% ULTIMATE FACTORED LOAD

FIG. 18: TEST OF C-102 DERWENT FIN

FIG. 19: TEST OF C-102 DERWENT TAILPLANE

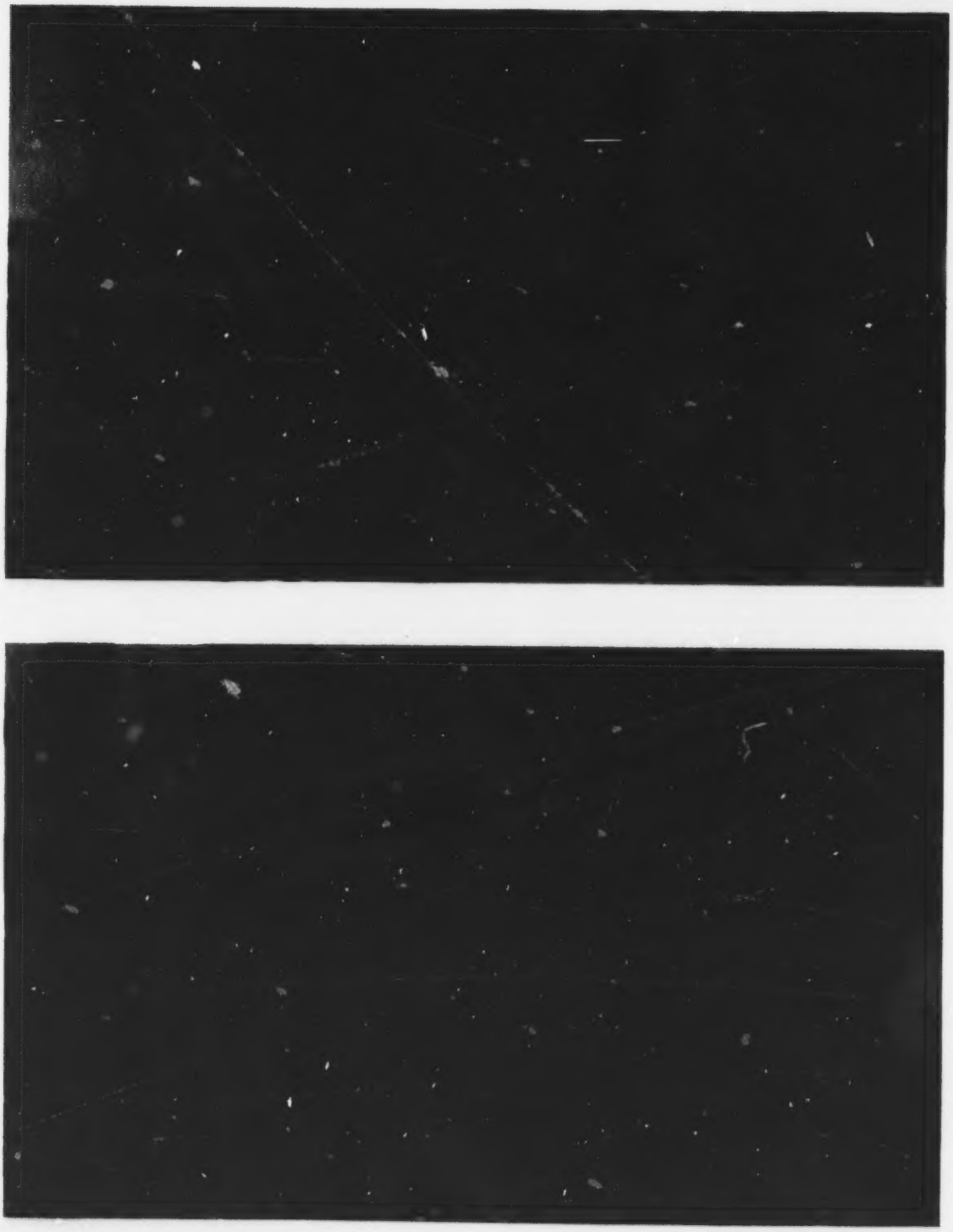

FIG. 20: TEST SET-UP FOR C-102 DERWENT MAINPLANE AND CENTRE FUSELAGE

FIG. 21: NRL 200 CHANNEL STRAIN RECORDER

FIG. 22: TEST OF C-100 TAILPLANE

FIG. 23: TEST SET-UP FOR C-100 OUTER WING SHOWING RIVETED ATTACHMENTS TO LEADING EDGE

FIG. 24: C-100 EMPENNAGE UNDER PROOF LOAD

FIG. 25: LOADING AT THE CENTRE ELEVATOR HINGE OF C-100 TAILPLANE SHOWING APPLICATION OF A COUPLE TO THE HINGE AND CLOSE-UP OF LOAD CELL

FIG. 26: AUTOMATED REPEATED LIMIT LOAD TEST ON C-100 MARK 3 OUTER WING

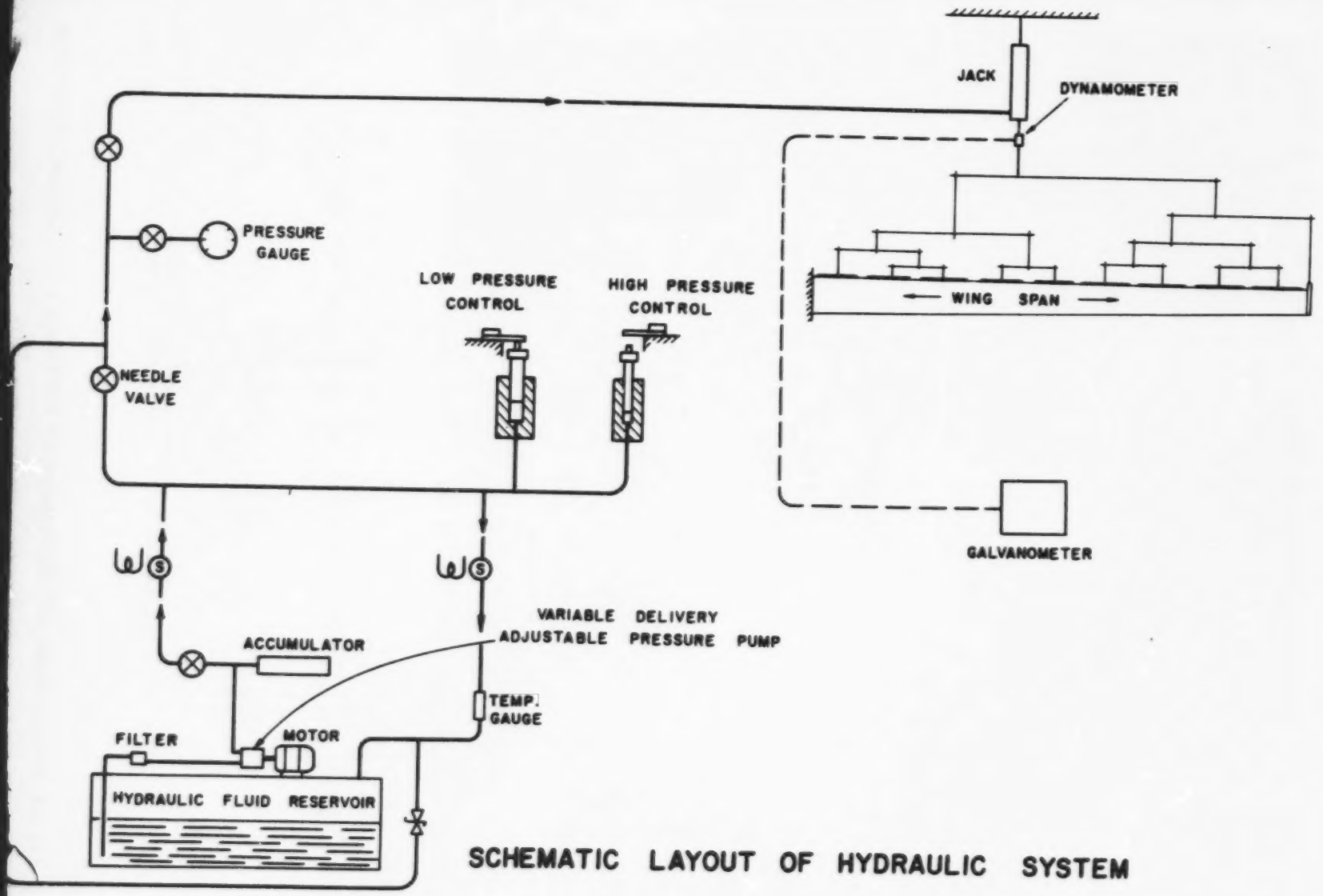

FIG. 27: AUTOMATIC LOADING SYSTEM FOR REPEATED LOAD TESTS

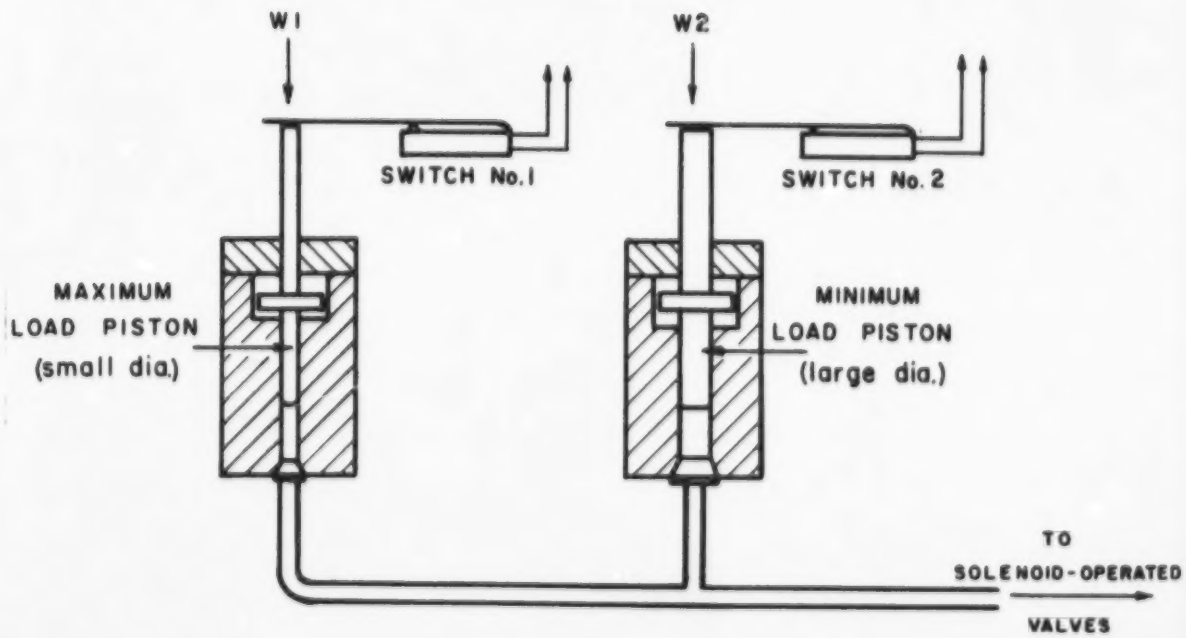

FIG. 28: PHOTOGRAPH AND SCHEMATIC OF REPEATED LOAD BALANCING SYSTEM

FIG. 29: STATIC TEST SET-UP FOR FOUND BROTHERS FBA-2 AIRFRAME

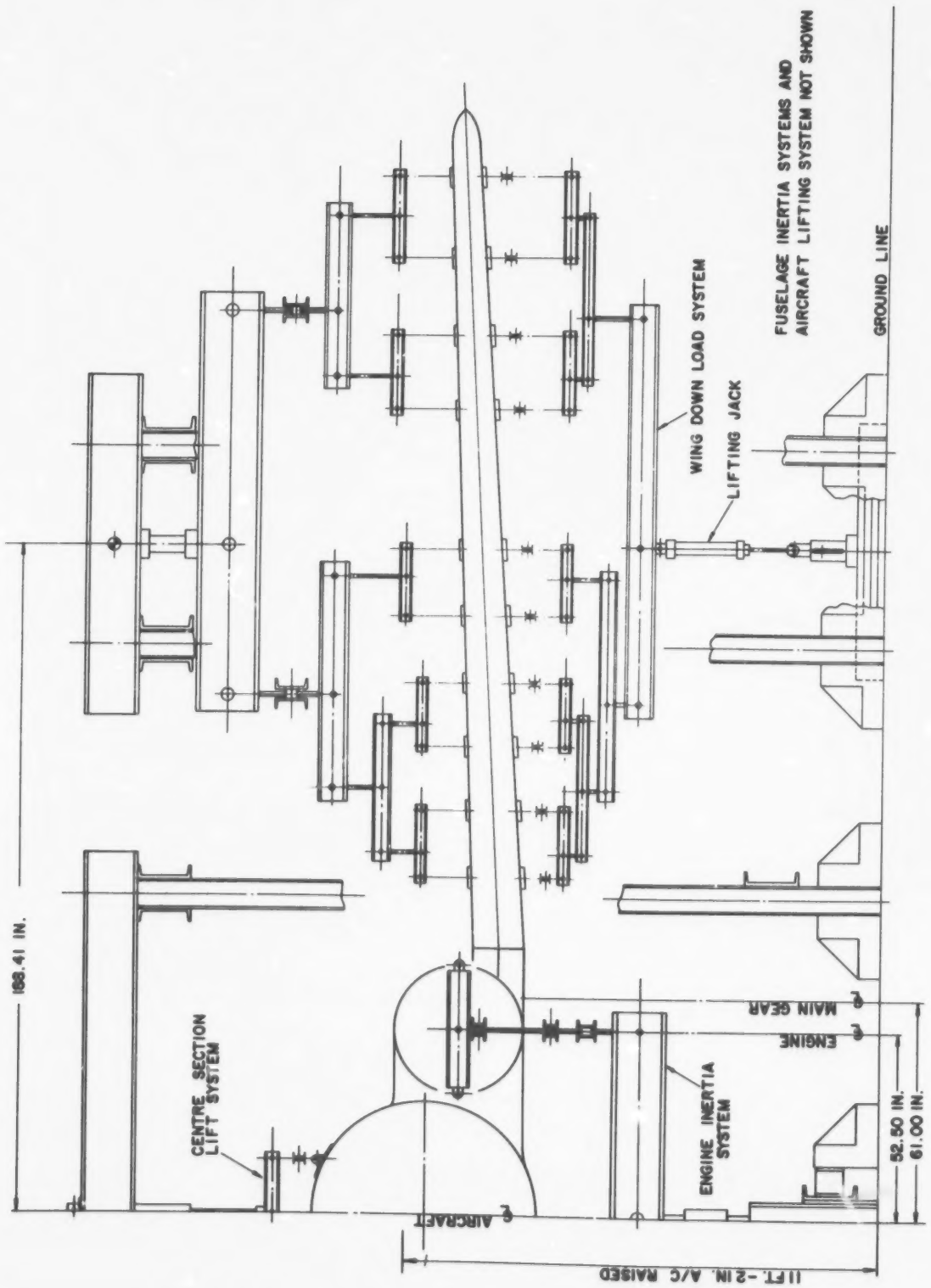

FIG. 30: C-100 WING-FUSELAGE TEST ARRANGEMENT – FRONT ELEVATION

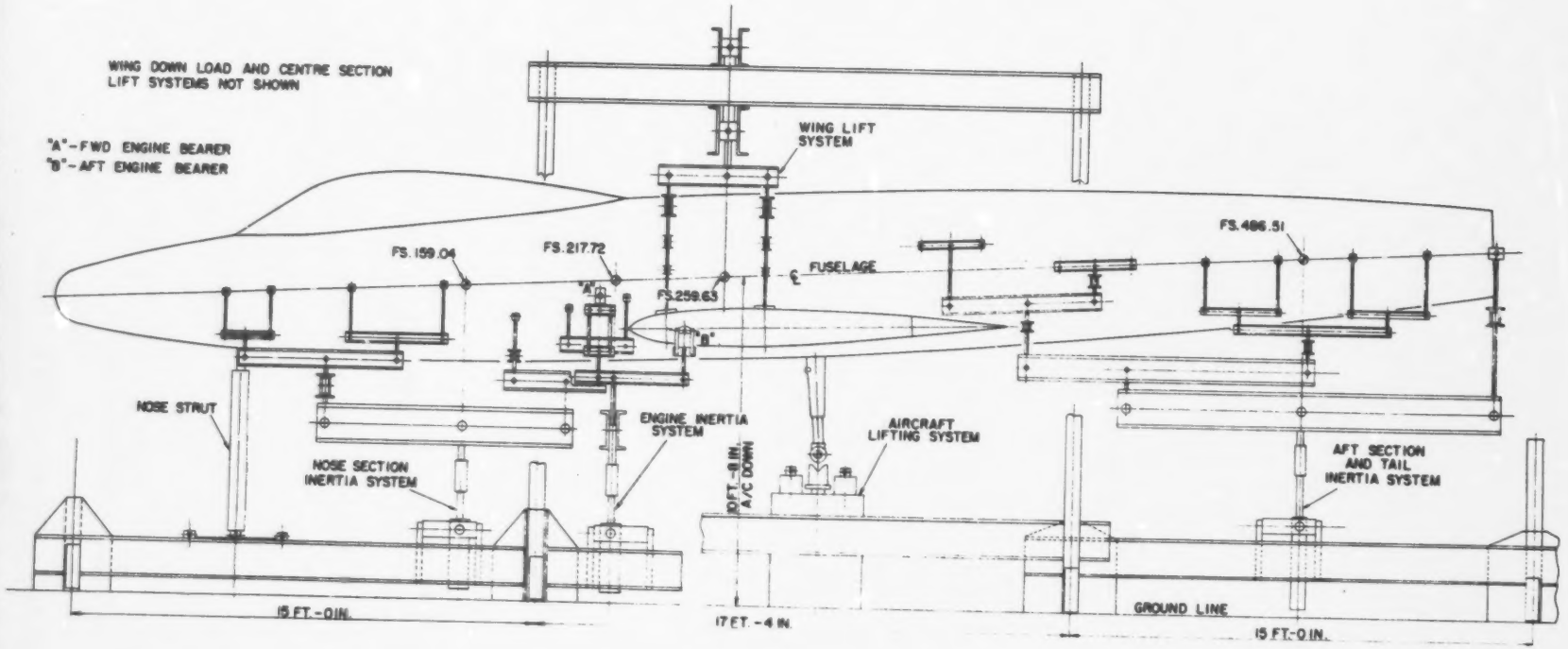

FIG. 31: C-100 WING-FUSELAGE TEST ARRANGEMENT – SIDE ELEVATION

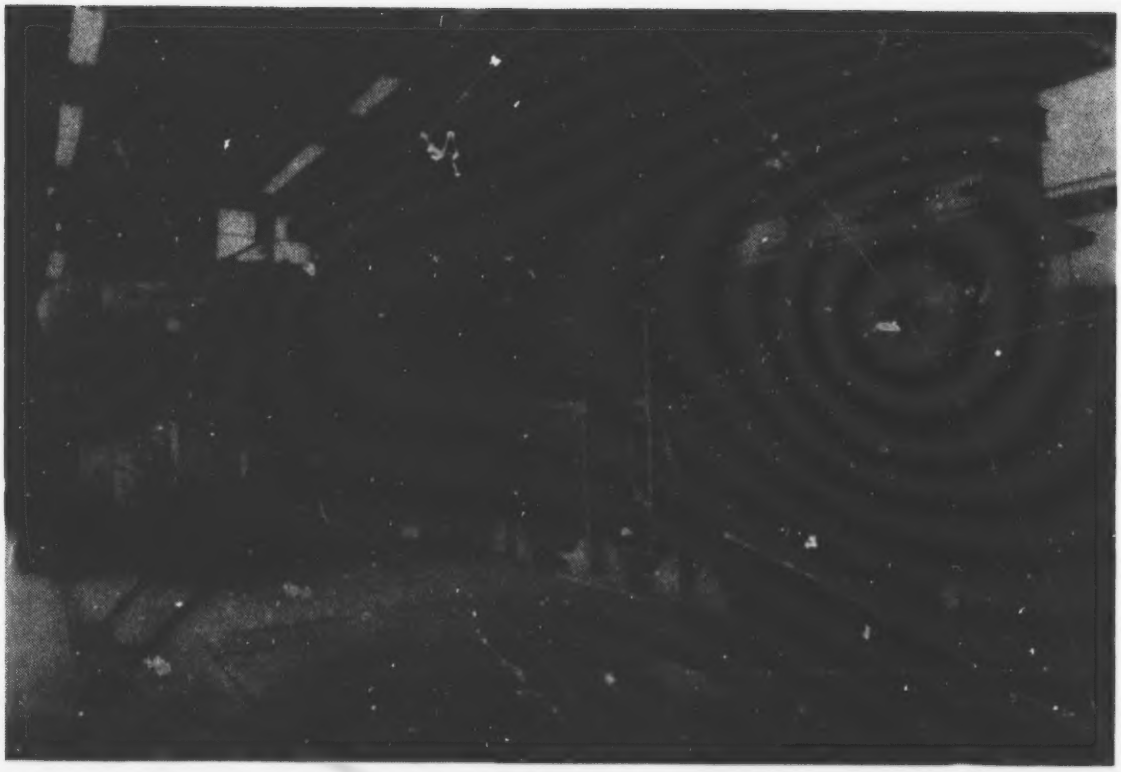

FIG. 32: C-100 WING-FUSELAGE FATIGUE TEST ARRANGEMENT

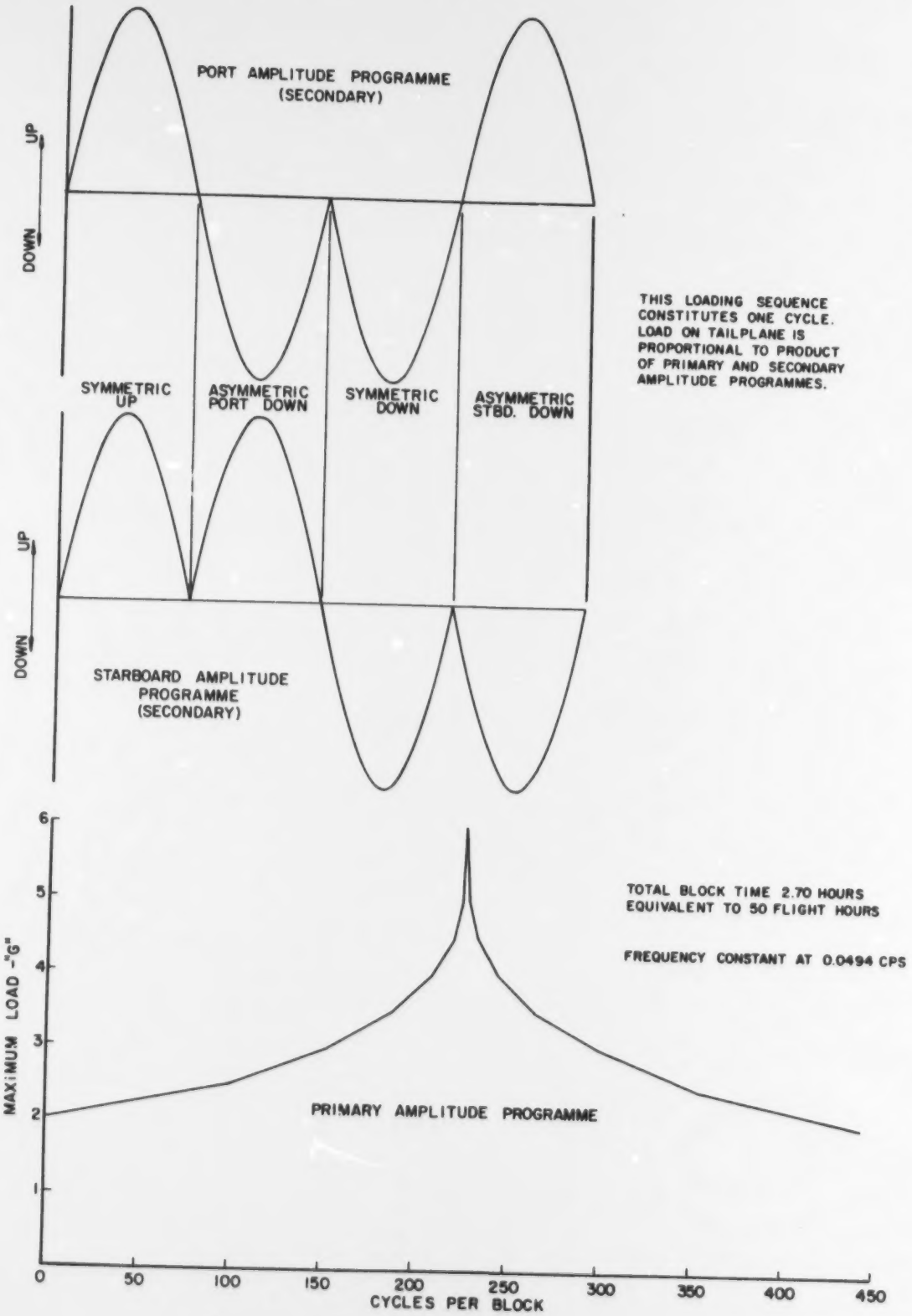

FIG. 33: TEST LOAD HISTORY FOR C-100 TAILPLANE

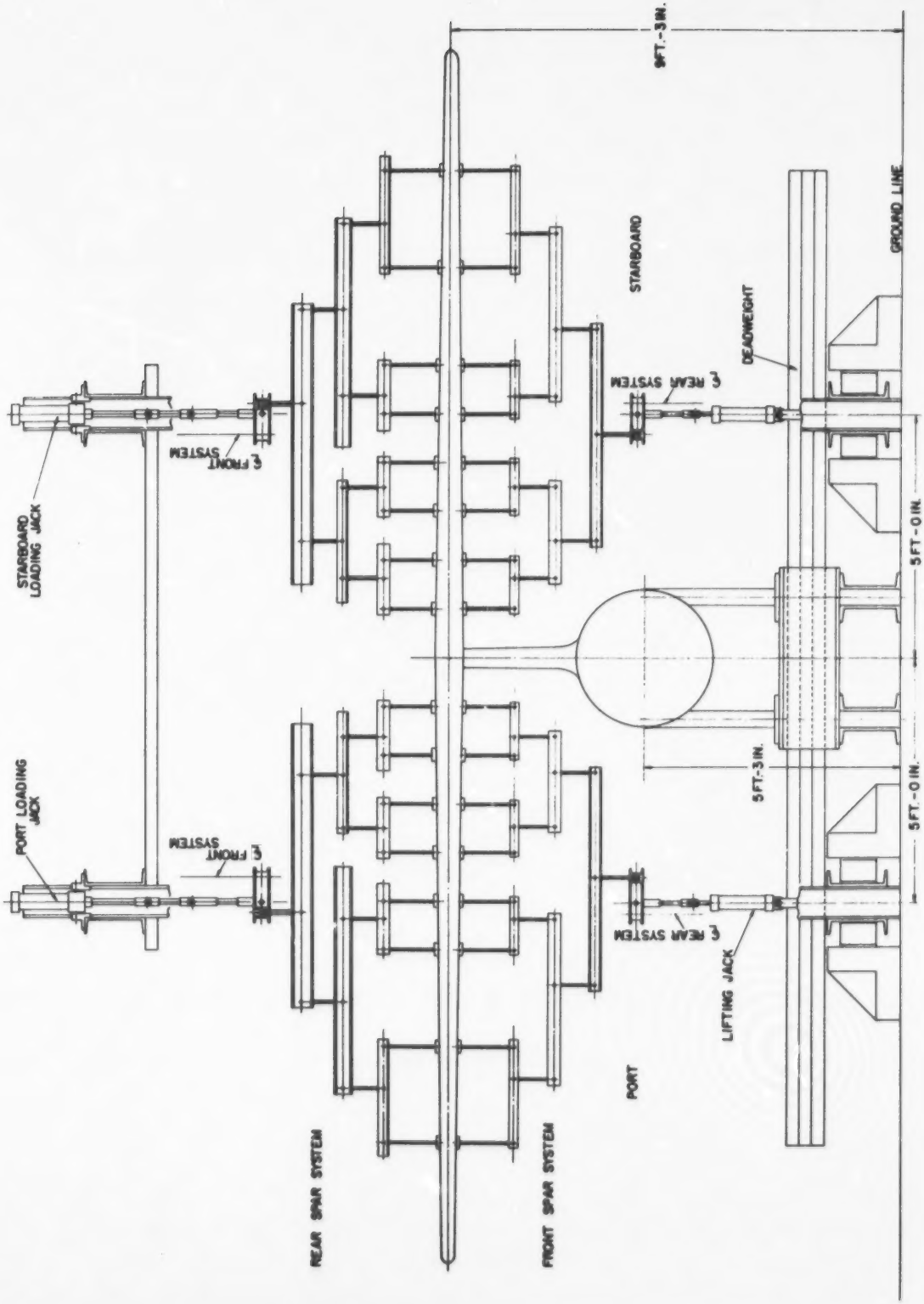

FIG. 34: C-100 TAIL UNIT TEST ARRANGEMENT – REAR ELEVATION

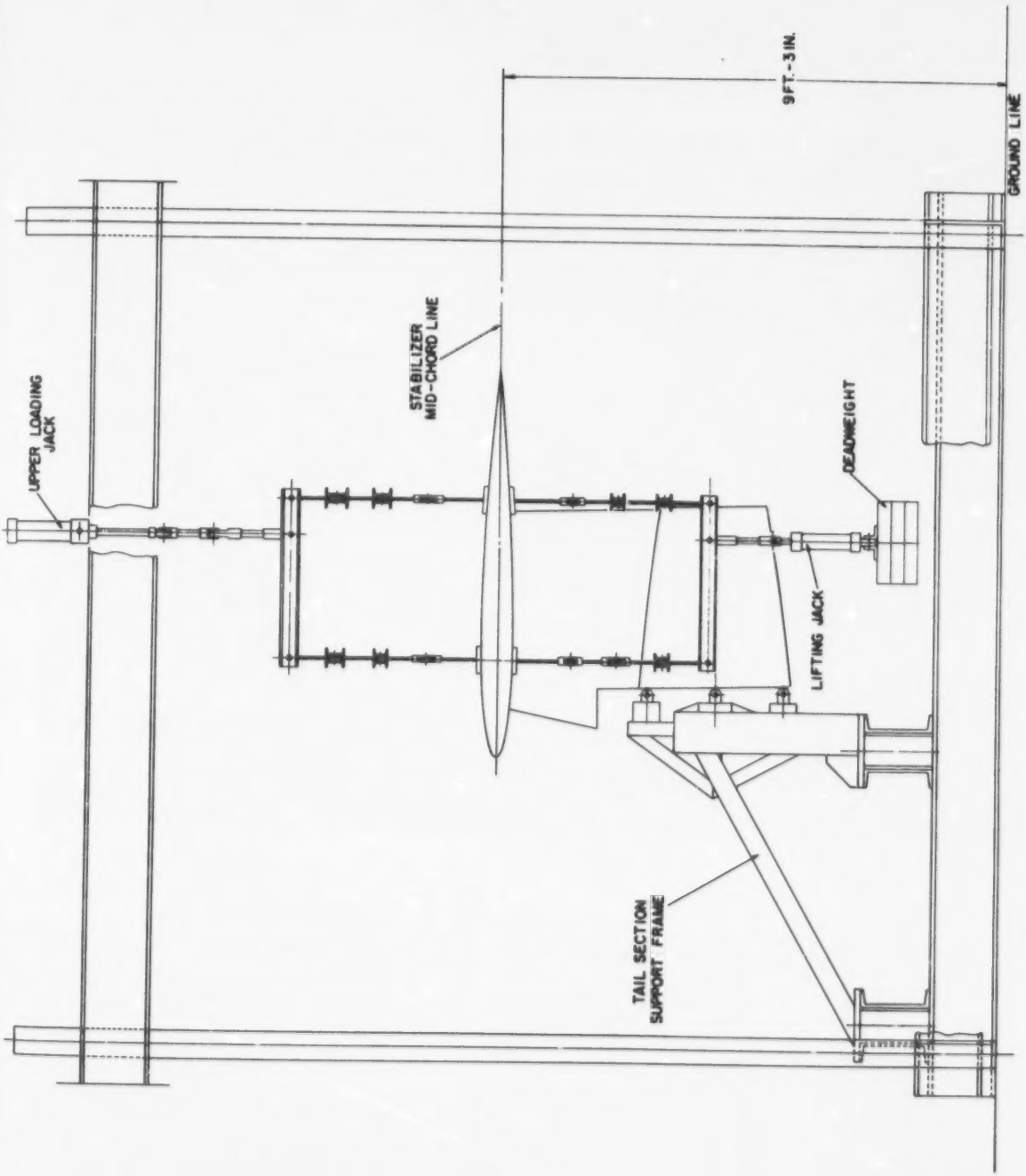

FIG. 35: C-100 TAIL UNIT TEST ARRANGEMENT – SIDE ELEVATION

FIG. 36: C-100 TAIL UNIT FATIGUE TEST ARRANGEMENT

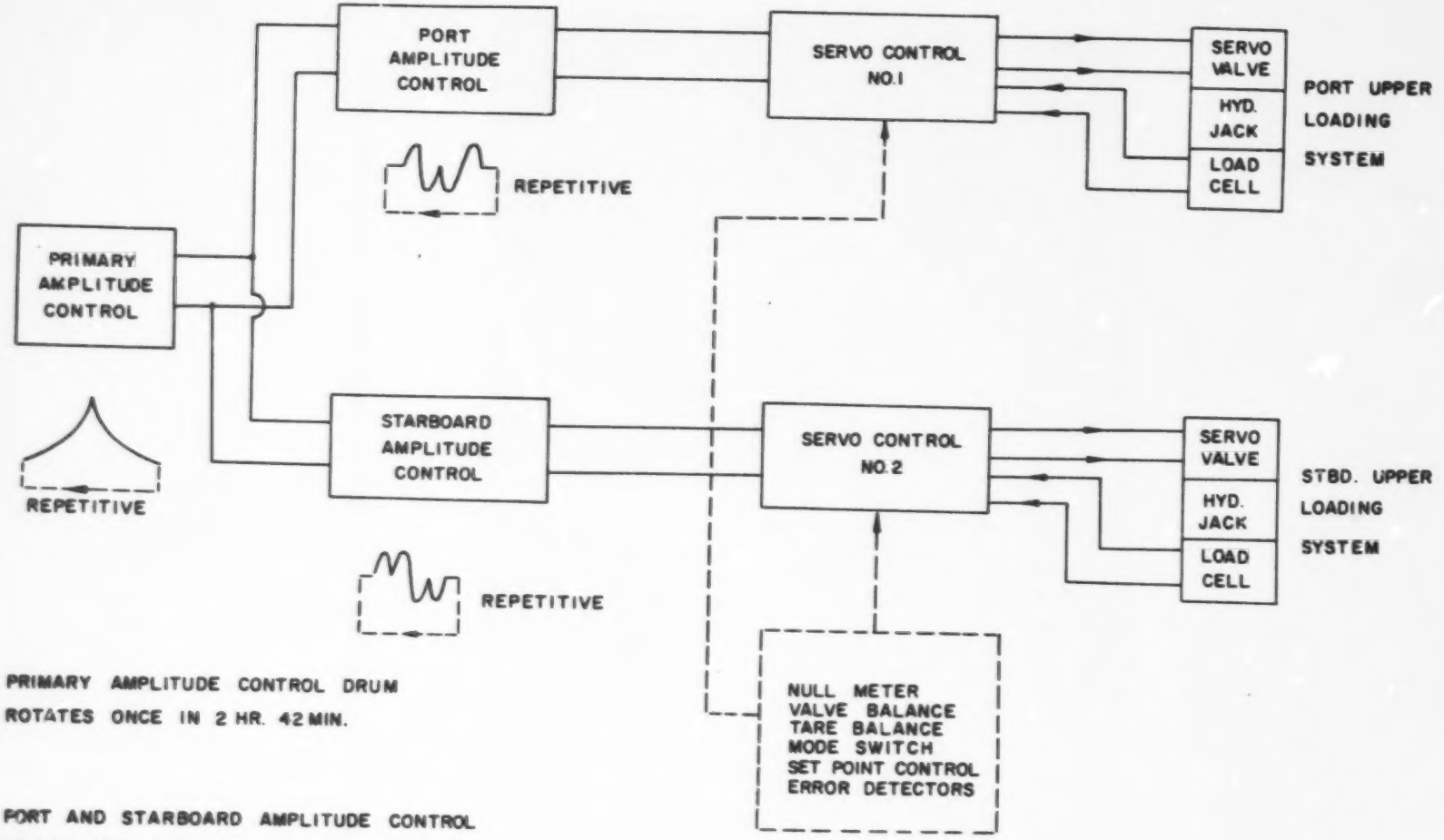

FIG. 37: BLOCK DIAGRAM OF LOAD CONTROL SYSTEM FOR C-100 TAILPLANE TEST

FIG. 38: VIEW OF TUTOR FATIGUE TEST SHOWING WING LOADING SYSTEM

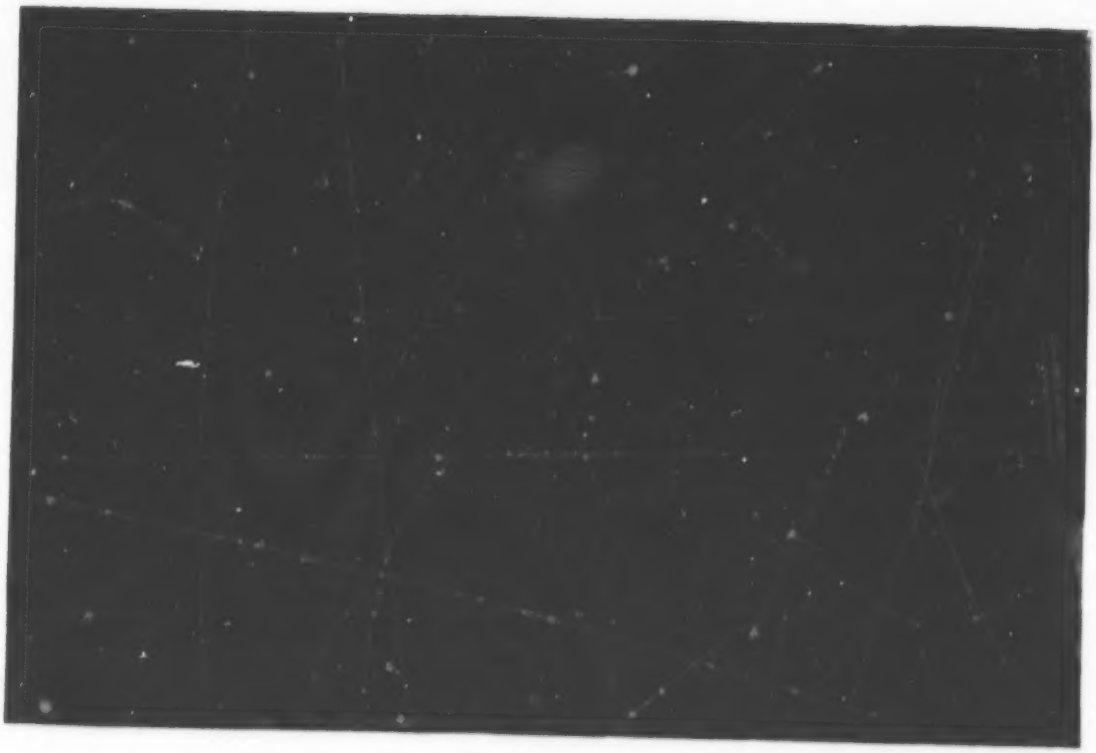

FIG. 39: FUSELAGE LOADING SYSTEM FOR TUTOR WING FATIGUE TEST

FIG. 40: TUTOR WING FATIGUE TEST SET-UP

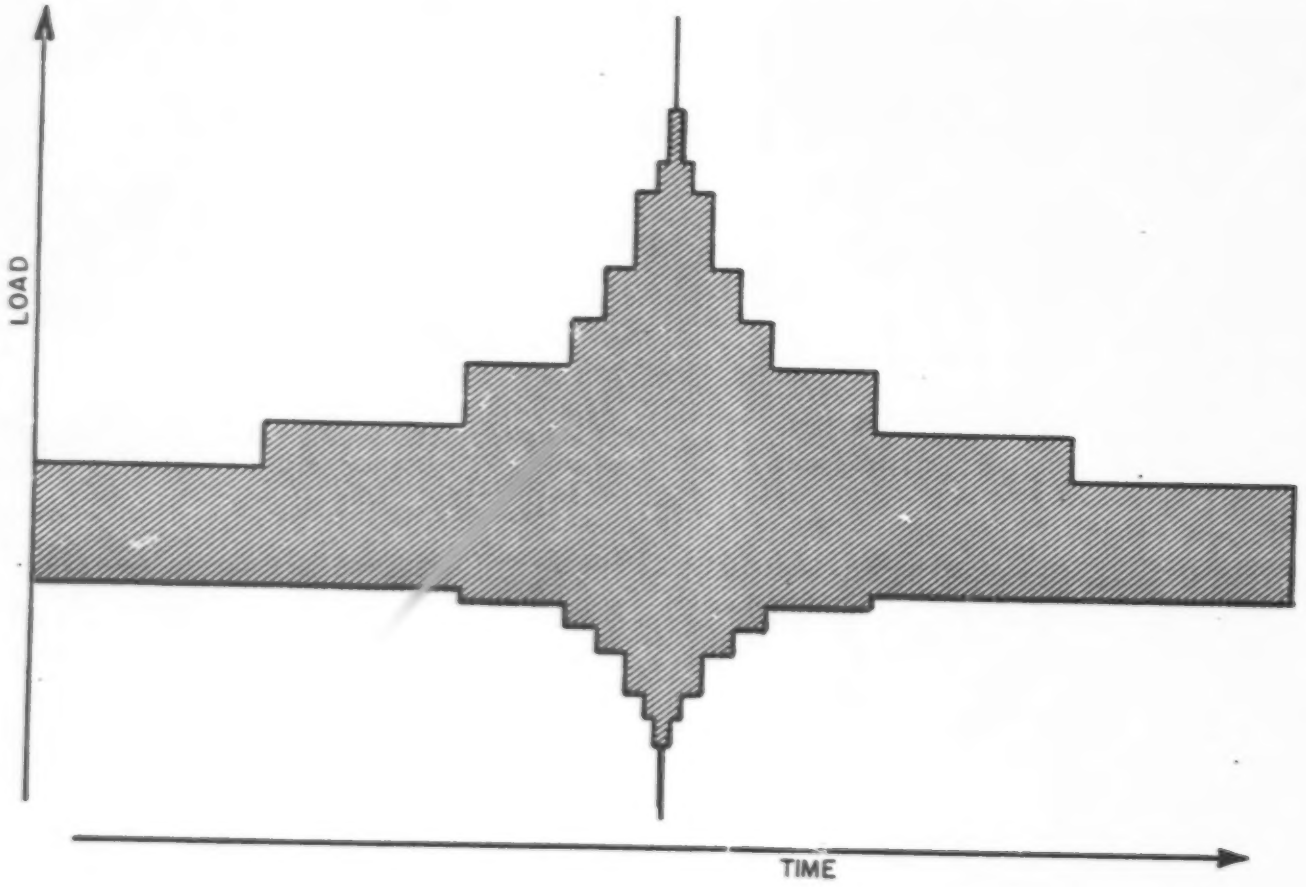

FIG. 41: NINE LEVEL LOW-HIGH-LOW LOADING SEQUENCE USED IN TUTOR FATIGUE TEST (TRAINER ROLE)

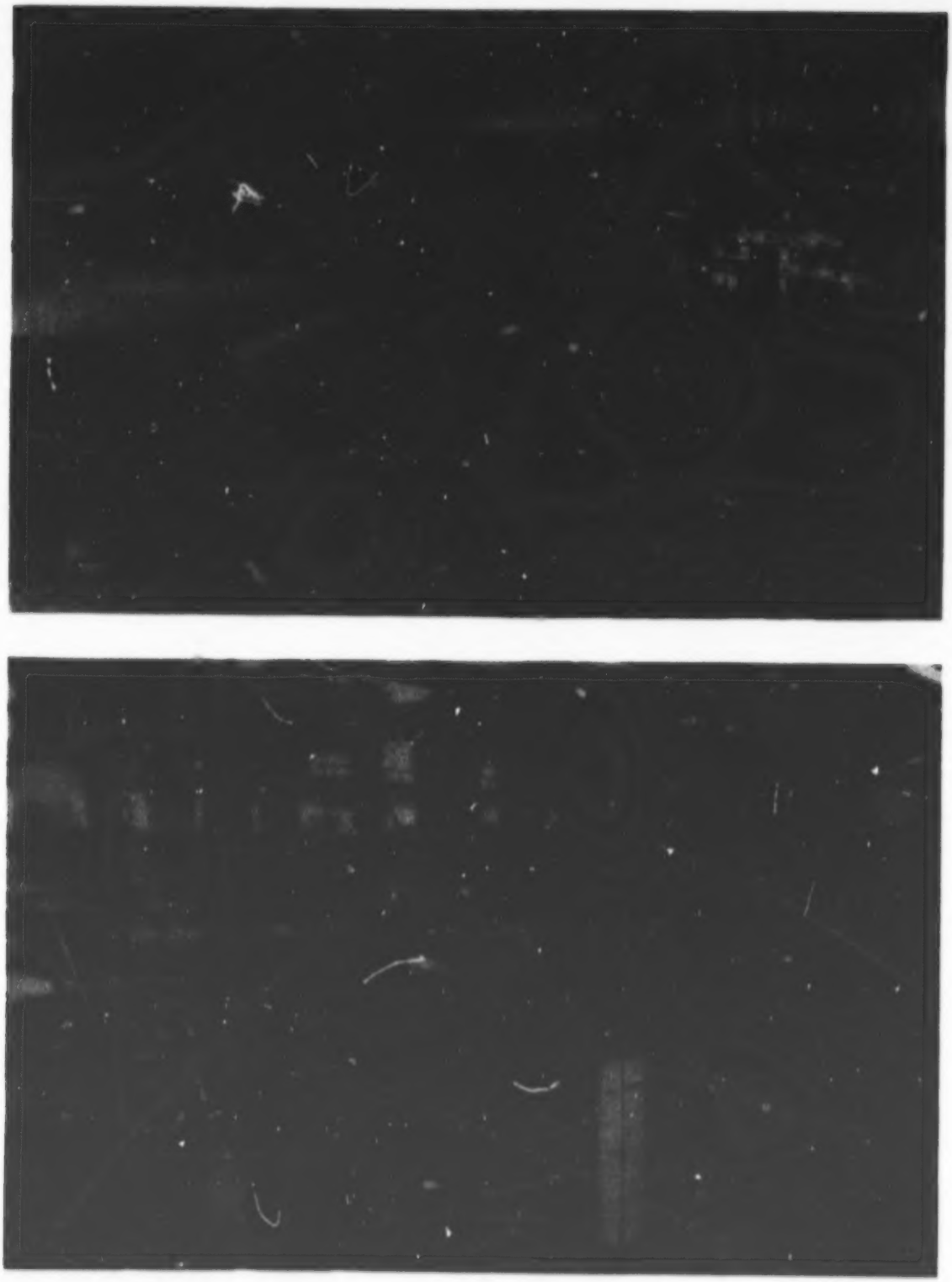

FIG. 42: TRACKER WING FATIGUE TEST SET-UP

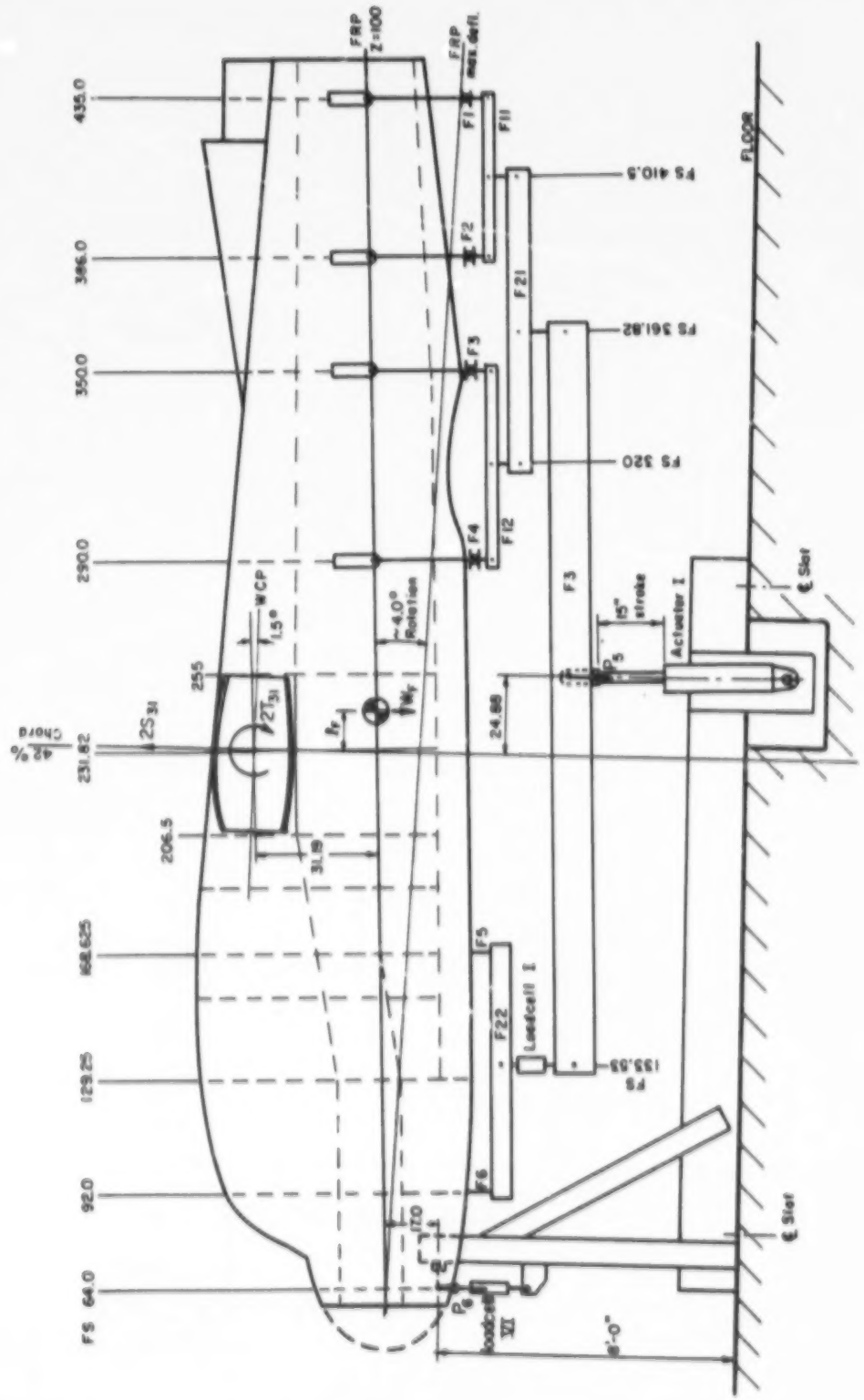

FIG. 43: UNDER FUSELAGE WHIFFLETREE SYSTEM

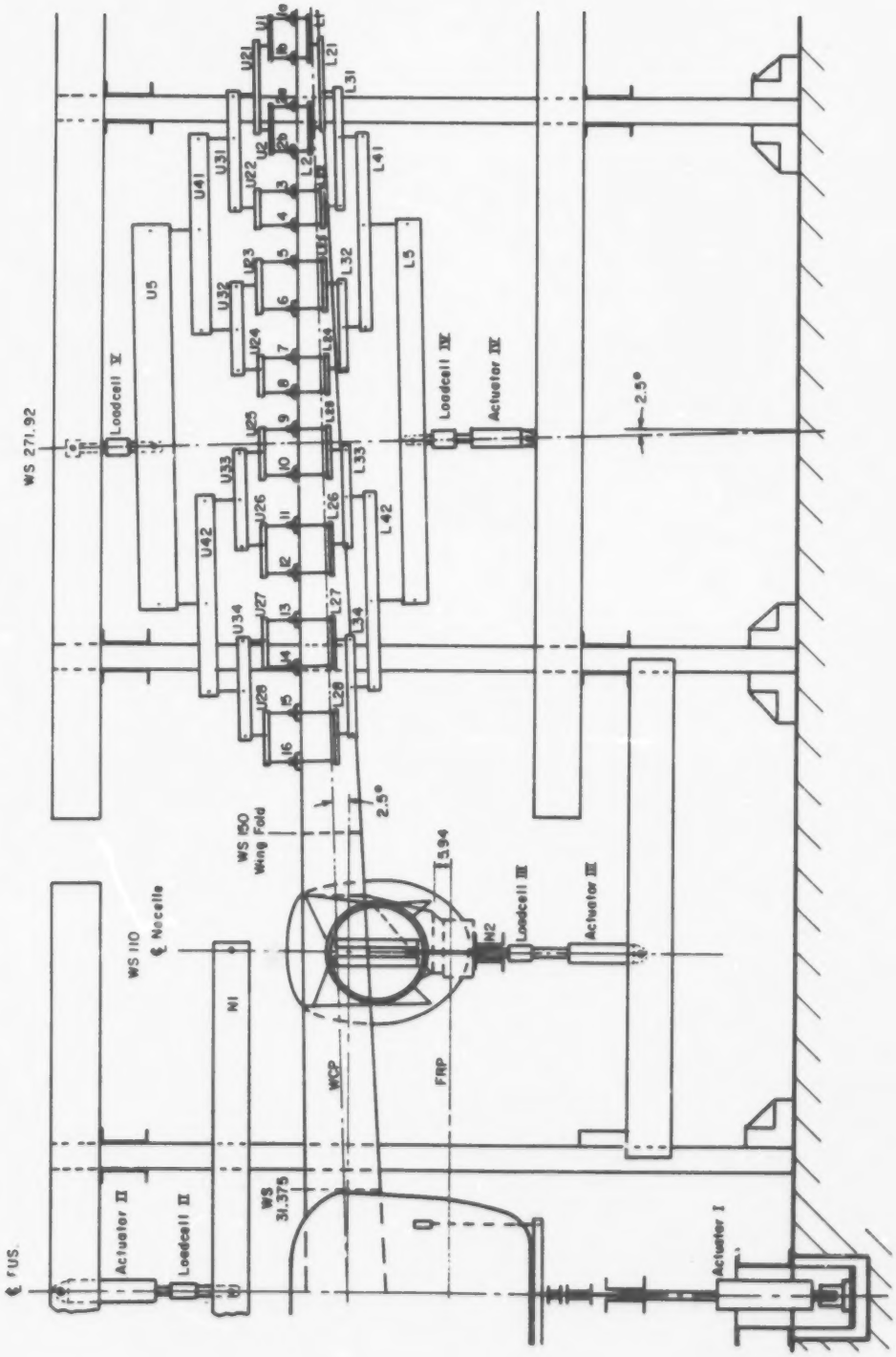

FIG. 44: TRACKER WING LOADING SYSTEM

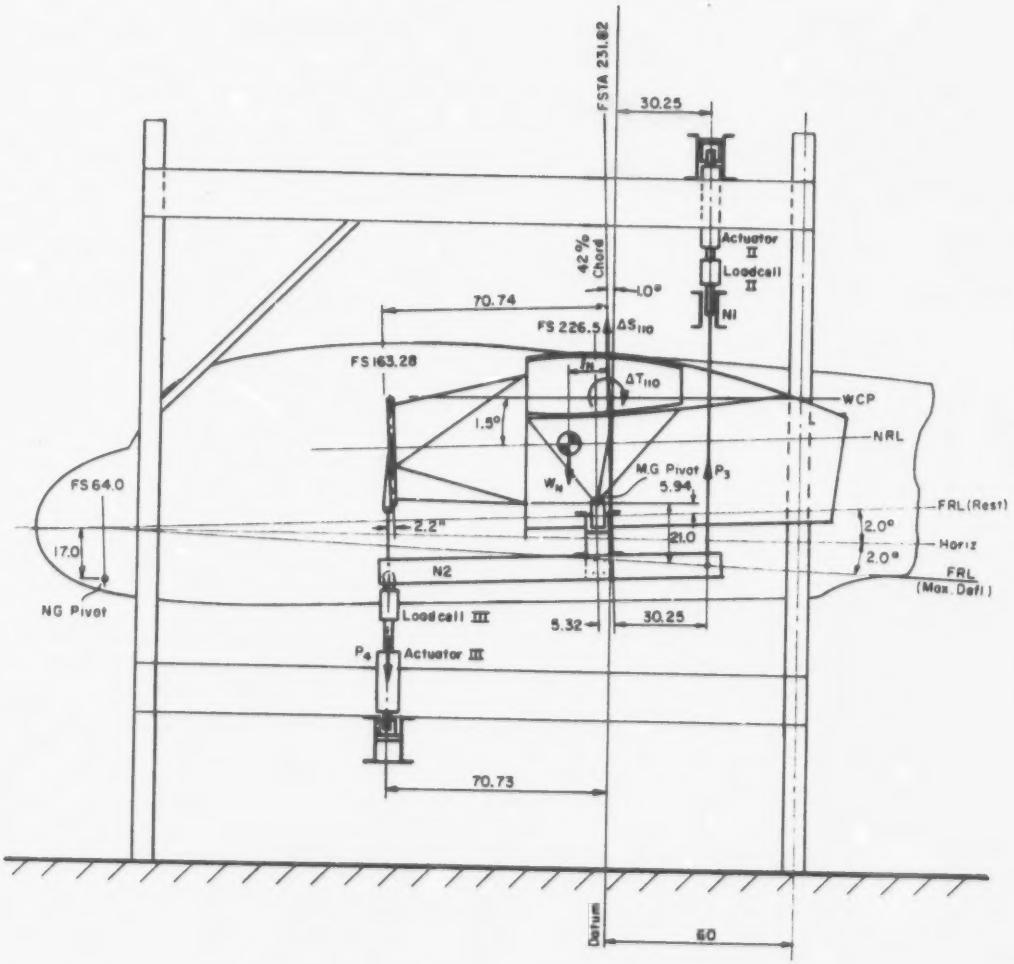

FIG. 45: NACELLE LOADING SYSTEM

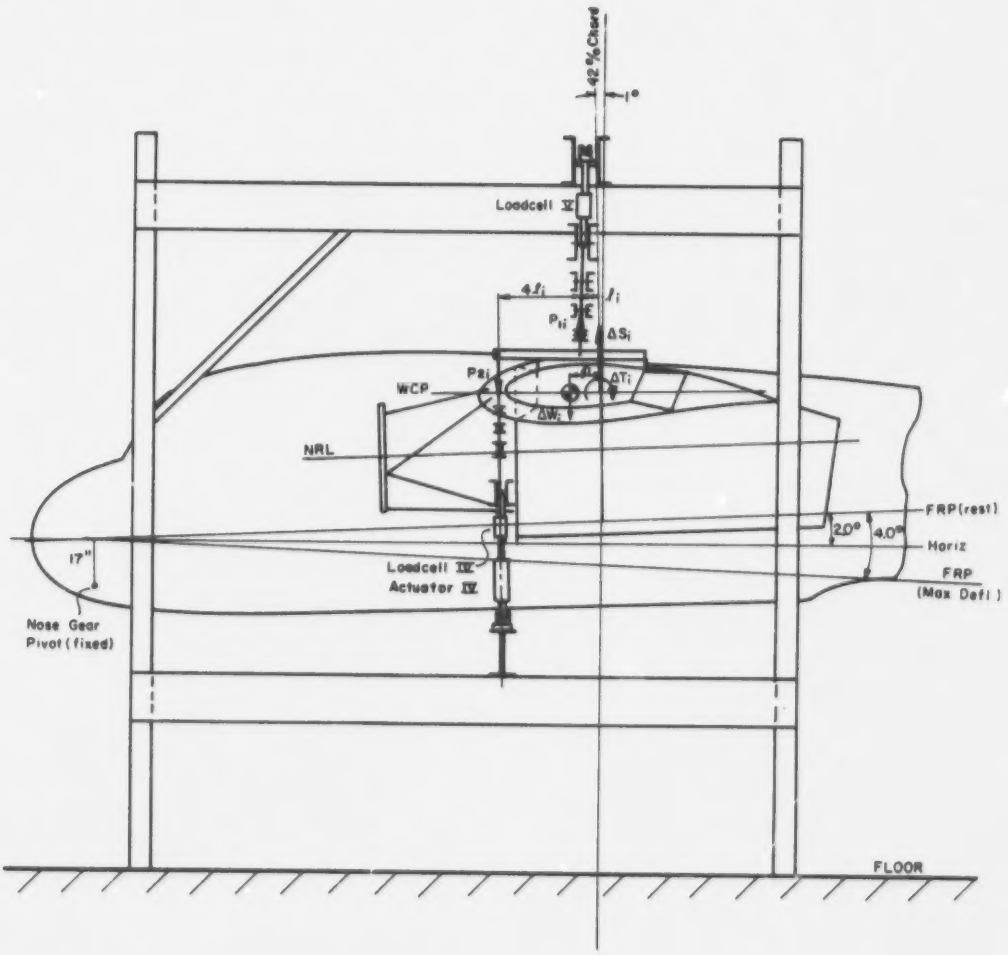

FIG. 46: OUTER WING LOADING SYSTEM

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SUMMARY/SOMMAIRE

This report presents an historical review of the work of the Structures and Materials Laboratory of NAE in the area of full scale testing of large aircraft structures. It covers the period from 1941 to 1984, starting with a static strength test of a moulded wood Anson fuselage and finishing with a fatigue test of a Grumman Tracker wing. Brief details of the loading arrangements and test results are included for each test component and these are used to trace the development of the laboratory from the use of rulers and shot bags to computer-controlled servo-hydraulic actuators.

(15)